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A METHOD FOR PREDICTING THE STATIC AERODYNAMIC CHARACTERISTICS OF TYPICAL
MISSILE CONFIGURATIONS FOR ANGLES OF ATTACK TO 180 DEGREES

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SUMMARY

A method for predicting the static, longitudinal aerodynamic characteristics of typical missile configurations at zero roll angle (i.e., in a plus configuration) has been developed and programmed for use on the IBM 7090 digital computer. It can be applied throughout the subsonic, transonic, and supersonic speed regimes to slender bodies of revolution or to nose-cylinder body combinations with low aspect-ratio lifting surfaces. The aerodynamic characteristics can be computed for missile configurations operating at angles of attack up to 180 degrees. The effect of control surface deflections for all modes of aerodynamic control are taken into account by this method. The method is based on well-known linear, nonlinear crossflow and slender body theories with empirical modifications to provide the high angle of attack capability. Comparisons of the theory with experimental data are presented to demonstrate the accuracy of the method.

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NOMENCLATURE

AR	exposed aspect ratio
b	semispan of an aerodynamic surface including the body radius, feet
C_D	total drag coefficient
C_{D_b}	base drag coefficient
C_{D_c}	crossflow drag coefficient
C_{D_f}	friction drag coefficient
C_{D_i}	induced drag coefficient
C_{D_o}	total zero-lift drag coefficient
C_{D_p}	pressure drag coefficient
C_{D_V}	wave drag coefficient
C_f	incompressible skin-friction coefficient
C_{f_c}	compressible skin-friction coefficient
$C_{D_{FP}}$	drag of a flat plate normal to the flow
C_L	total lift coefficient
C_{L_α}	lift curve slope, per radian
C_m	total longitudinal pitching moment coefficient
C_r	root-chord of an aerodynamic surface, feet
C_t	tip-chord of an aerodynamic surface, feet
d	diameter of the body at any station, feet
d_B	base diameter of the body, feet
d_N	diameter of the nose at the nose-body juncture, feet
f	spanwise location of the vortex which emanates from the forward surface, feet
h_A	height of the trailing vortex above the body centerline at the aft surface center of pressure, feet
i	downwash interference constant

NOMENCLATURE

(continued)

$(k_2 - k_1)$	apparent mass factor
K	linear lift interference factor due to angle of attack
K'	linear lift interference factor due to control surface deflection
l_B	total length of the body, feet
l_N	length of the nose, feet
l_{REF}	arbitrary reference length, usually the maximum body diameter, feet
l_T	distance from the tip of the nose to the intersection of the tail leading edge with the body, feet
l_W	distance from the tip of the nose to the intersection of the wing leading edge with the body, feet
M	free-stream Mach number
m	cotangent of the leading edge sweep angle
r	radius of the body at any station, feet
Re	Reynolds number
S_B	base area of the body, (feet) ²
S_F	exposed planform area of one pair of forward lifting surfaces, (feet) ²
S_N	body cross-sectional area at the nose, body juncture, (feet) ²
S_P	planform area of the body, (feet) ²
S_s	surface area of the body, (feet) ²
S_T	exposed planform area of one pair of tail surfaces, (feet) ²
S_T'	planform area of one pair of tail surfaces as obtained by extending the leading and trailing edges to the centerline of the body, (feet) ² . See Figure 2.

NOMENCLATURE

(continued)

S_W	exposed planform area of one pair of wings, (feet) ²
S'_W	planform area of one pair of wings as obtained by extending the leading and trailing edges to the centerline of the body, (feet) ² . See Figure 2.
V_B	volume of the body, (feet) ³
X_{CG}	distance from the nose to the missile center of gravity, (feet) ²
\bar{X}	distance to the surface center of pressure as measured from the intersection of the leading edge of the aerodynamic surface with the body, feet
X_{CP}	distance from the nose to the center of pressure location, feet
X_h	distance from the intersection of the panel leading edge and the body to the hinge line, feet
X_p	distance from the nose to the centroid of the body planform area, feet
α	missile angle of attack, degrees
β	compressibility factor, $\sqrt{M^2-1}$
δ	control surface deflection, degrees (See Figure 1 for sign conventions)
ΔC_D	component of the induced drag coefficient
ΔC_{D_0}	increment of wave drag for the transonic speed regime
η	ratio of the drag coefficient of a circular cylinder of finite length to that of infinite length
θ_N	conical nose semi-vortex angle, degrees
λ	lifting surface taper ratio, C_t/C_r
Λ	leading edge sweep angle, degrees
$\Lambda_{c/4}$	sweep angle of the quarter chord line, degrees

NOMENCLATURE

(continued)

SUBSCRIPTS

A	aft lifting surface, alone
B	body alone
BT	body in the presence of the tail
BT - α	body in the presence of the tail due to angle of attack
BT - δ	body in the presence of the tail due to control surface deflection
BW	body in the presence of the wing
BW - α	body in the presence of the wing due to angle of attack
F	forward surface alone
FB	forward surface in the presence of the body
N	nose
T	tail alone
T - α	tail alone due to angle of attack
T - δ	tail alone due to control surface deflection
TB	tail in the presence of the body
TB - α	tail in the presence of the body due to angle of attack
TB - δ	tail in the presence of the body due to control surface deflection
TV	tail, nonlinear component
W	wing alone
WB	wing in the presence of the body
WB - α	wing in the presence of the body due to angle of attack
WV	wing, nonlinear component

The control surface is defined as the tail regardless of the mode of control; the fixed surface is defined as the wing (see Figure 1).

INTRODUCTION

Increasing maneuverability requirements of missiles indicated a need for predicting the aerodynamic characteristics, including lift, drag, and pitching moment, of missile configurations to angles of attack of 90 degrees and higher. A study showed that existing methods for computing these aerodynamic characteristics are based on a number of different theories all of which are applicable only to small angles of attack. To fulfill the high angle of attack requirements, a method for determining the aerodynamic characteristics of low aspect-ratio configurations at zero roll angles operating at angles of attack up to 180 degrees has been developed. The method is applicable throughout the subsonic, transonic, and supersonic speed regimes up to $\beta \cdot AR \approx 10.0$, and accounts for control surface deflections.

The method is composed of well-known linear, nonlinear crossflow, and slender body theories which have been modified to provide the required high angle of attack capability. These theories can be applied to slender bodies of revolution or nose-cylinder bodies with canard, wing, or tail controls (Figure 1).

This report describes the methods developed and the computer program which has been written for use on the IBM 7090 digital computer. The description of the method is divided into three parts: lift, drag, and pitching moment. For the sake of clarity, the description of the method is kept to a minimum, without lengthy justification and description of the techniques employed. The reader is referred to the references for detailed descriptions of the various theories. The description of the computer program consists of a brief discussion of the main program and subroutines and complete instructions required for use of the program. Comparisons of theoretical results with experimental data are presented for angles of attack up to 90 degrees over the entire speed range to demonstrate the accuracy of the theories. Some data for a missile configuration at 180 degrees angle of attack is available and is compared with the theoretical results.

LIFT CHARACTERISTICS

The total lift on the missile is the sum of the body lift, the lift due to the aerodynamic surfaces, and the interference lift between the forward and aft surfaces. The lift on the body and aerodynamic surfaces is composed of two components: linear lift including the effects of the body-lifting surface interaction and nonlinear crossflow lift. In general, the crossflow lift component is caused by flow separation which occurs at angle of attack, while the interference component is the lift-loss on the aft lifting surface due to downwash from the forward surface (Reference 1).

Allen, References 2 and 3, developed a method for predicting the total lift on bodies of revolution at angles of attack. This method includes the linear or potential flow component and two nonlinear components: the viscous crossflow force and the viscous axial force. Because the contribution of the axial force component to the body lift is small, it is usually neglected. Allen's expression for the body lift is

$$C_{L_B} = (k_2 - k_1) \left(\frac{S_B}{S_{REF}} \right) \sin 2\alpha \cos \frac{\alpha}{2} + n C_{dc} \left(\frac{S_P}{S_{REF}} \right) \sin^2 \alpha \cos \alpha - C_{D_B} \cos^2 \alpha \sin \alpha \quad (1)$$

where the first term is the linear contribution and the second term is the nonlinear contribution. The apparent mass factor, $k_2 - k_1$, and the drag ratio, n , can be obtained from Figure 3, while the crossflow drag coefficient, C_{dc} , is obtained from Figure 4. Comparisons of theory with experimental data for numerous bodies of revolution over a wide range of Mach numbers and angles of attack are presented in Reference 3. It should be noted that although this expression for the lift is independent of the nose shape, good agreement with experiment is indicated in Reference 4 for a body with an unusual shape.

The linear lift characteristics of low aspect-ratio lifting surfaces whose cross-sections are thin and symmetrical are generally a function of speed, planform area, and aspect-ratio. When the diameter of the missile body is of the same order of magnitude as the span of the

lifting surfaces, the effects of body-wing and body-tail interactions are significant. Hence, the linear lift of the aerodynamic surfaces is composed of two components: the lift on the surface in the presence of the body, and the added lift on the body due to the presence of a surface. Most low aspect-ratio missile configurations exhibit a nonlinear dependence of lift on angle of attack, especially at the higher angles. One primary cause of this is the crossflow lift component which is due to lateral flow separation and the formation of free vortices on the upper surface. This nonlinear dependence is analyzed in References 1 and 5 and summarized by Eaton in Reference 6.

The expression for the total wing lift based on an arbitrary reference area is

$$C_{L_W} = C_{L_{WB-\alpha}} + C_{L_{BW-\alpha}} + C_{L_{WV}} \quad (2)$$

In order to provide high angle of attack capability, it is necessary to modify both the linear and nonlinear theories. The lift on the wing in the presence of the body, as presented in Reference 2, is a linear function of angle of attack and can be expressed as

$$C_{L_{WB-\alpha}} = K_{WB} C_{L_{\alpha_W}} \left(\frac{S_W}{S_{REF}} \right) \alpha \quad (3)$$

Since the lift force does not vary linearly with angle of attack at high angles, Equation (3) is modified such that the linear lift becomes a function of $\sin \alpha$ as shown below

$$C_{L_{WB-\alpha}} = K_{WB} C_{L_{\alpha_W}} \left(\frac{S_W}{S_{REF}} \right) \sin \alpha \quad (4)$$

This component of the linear lift is modified further to satisfy the end condition of zero lift at 90 degrees angle of attack. The resulting expression for the linear wing lift in the pressure of a body is

$$C_{L_{WB-\alpha}} = K_{WB} C_{L_{\alpha_W}} \left(\frac{S_W}{S_{REF}} \right) \sin \alpha \cos \alpha \quad (5)$$

It is important to note that for small angles of attack the modified theory should be very close to the method of Reference 1 since $\sin \alpha \approx \alpha$ and $\cos \alpha \approx 1$. Similarly, the additional lift on the body due to the presence of the wing is

$$C_{L_{BW-\alpha}} = K_{BW} C_{L_{\alpha_W}} \left(\frac{S_W}{S_{REF}} \right) \sin \alpha \cos \alpha \quad (6)$$

The parameters, K_{WB} and K_{BW} , are determined from Figure 5. $C_{L_{\alpha_W}}$ is obtained from Figure 6 by multiplying $\frac{C_{L_{\alpha_W}}}{AR}$ by the aspect-ratio

if $AR < 1.0$. When the aspect ratio is greater than one, the lift-curve slope is obtained from the following equation

$$C_{L_{\alpha_W}} = \left(\frac{1}{AR^{(AR-1)/AR}} \right) \left(\frac{C_{L_{\alpha_W}}}{AR} \right) AR \quad (7)$$

where $\left(\frac{C_{L_{\alpha_W}}}{AR} \right)$ is obtained from Figure 6. The first term of Equation (7) is an empirical modification of the lift curve slope for a lifting surface with aspect-ratio greater than 1.

The nonlinear wing lift from Reference 6 is

$$C_{L_{WV}} = C_{dc} \sin^2 \alpha \left(\frac{S_w}{S_{REF}} \right) \quad (8)$$

This expression is modified to satisfy the aforementioned end condition with the result being

$$C_{L_{WV}} = C_{dc} \sin^2 \alpha \left(\frac{S_w}{S_{REF}} \right) \cos \alpha \quad (9)$$

where C_{dc} is obtained from Figure 7. It should be noted that the cross-flow drag coefficient is not appreciably affected by Mach number, (References 6 and 7); hence, C_{dc} is presented independent of Mach number.

The total tail lift is computed basically the same as the wing lift except for the lift due to deflection of the control surfaces. The tail lift is expressed as

$$C_{L_T} = C_{L_{TB-\alpha}} + C_{L_{BT-\alpha}} + C_{L_{TB-\delta}} + C_{L_{BT-\delta}} + C_{L_{TV}} \quad (10)$$

where $C_{L_{TB-\alpha}}$ and $C_{L_{BT-\alpha}}$ are obtained by applying Equations (5) and (6) to the tail surface. The other three components are:

$$C_{L_{TB-\delta}} = K_{TB} C_{L_{\alpha_T}} \sin \delta \left(\frac{S_T}{S_{REF}} \right) \cos(\alpha + \delta) \quad (11)$$

$$C_{L_{BT-\delta}} = K_{BT} C_{L_{\alpha_T}} \sin \delta \left(\frac{S_T}{S_{REF}} \right) \cos(\alpha + \delta) \quad (12)$$

$$C_{L_{TV}} = C_{dc} \sin^2(\alpha + \delta) \left(\frac{S_T}{S_{REF}} \right) \cos(\alpha + \delta) \quad (13)$$

The parameters, K_{BT}' and K_{TB}' , are obtained from Figure 5, while C_{dc} and $C_{L_{\alpha_T}}$ are obtained as specified for the wing. Notice that the nonlinear

lift is based on the local angle of attack, $(\alpha + \delta)$, of the control surface.

The lift-loss on the aft surface due to downwash from the forward surface is obtained from the method presented in Reference 1 and discussed in Reference 7. Since the method for computing this component of the total lift on the missile is both complex and lengthy, only the equations necessary to compute the lift-loss are presented. The reader is referred to Reference 1 for a detailed discussion of the assumptions and technique used in deriving the method. It is noted that the nomenclature used to describe the lifting surfaces is changed from wing and tail to forward and aft surfaces. This is necessary because the control surface, whether it be wing, canard, or tail type of control, is designated the tail and because the aft surface, regardless of the mode of control, is the one which is affected by downwash.

This method is valid for the entire speed range. The lift-loss due to downwash is

$$C_{L_i} = \frac{C_{L_{\alpha_F}} C_{L_{\alpha_A}} \left[K_{FB} \sin \alpha + K_{FB}' \sin \delta_F \right] i(b - r)_A S_F}{2\pi (AR)_A (f_F - r_F) S_{REF}} \quad (14)$$

This equation is obtained from line-vortex theory assuming only one trailing vortex per forward panel exists (see Figure 8). The lateral location, f_F , and the vertical location, h_A , of the vortex are required for use in Equation (14) and to compute the interference factor, i . The lateral location of the vortex on the forward surface expressed as a fraction of the exposed semispan of this surface is

$$\left(\frac{f - r}{b - r} \right)_F = \frac{\frac{\pi}{4} - \frac{\pi}{4} (r/b)_F^2 - (r/b)_F + \frac{\left[1 + (r/b)_F^2 \right]^2}{2 \left[1 - (r/b)_F^2 \right]} \sin^{-1} \left[\frac{1 - (r/b)_F^2}{1 + (r/b)_F^2} \right]}{2 \left[1 - (r/b)_F \right]} \quad (15)$$

For convenience, the right hand side of this equation is defined as A' . Isolating f_F results in the following expression

$$f_F = A'(b - r)_F + r_F \quad (16)$$

where f_F is the spanwise location of the vortex at the forward surface. Since the lateral location of the vortex with respect to the body axis is unchanged, the subscript may be dropped and Equation (16) may be written as

$$f = A'(b - r)_F + r_F \quad (17)$$

The vertical location of the vortex, h_A , is measured normal to the body axis at the center of pressure of the aft surface. The expression for h_A is

$$h_A = (c_r - x_h)_F \sin \delta_F + \left[l_A + (\bar{x}_{CP})_A - l_F - (c_r)_F \right] \sin \alpha \quad (18)$$

Note that the vertical location of the vortex is a function of both angle of attack and the deflection angle of the forward surface.

The interference factor, i , is given by

$$i = \left(\frac{2}{1+\lambda} \right) \left[L\left(\frac{\lambda}{b}, \frac{r}{b}, \frac{f}{b}, \frac{h}{b} \right) - L\left(\frac{\lambda}{b}, \frac{r}{b}, -\frac{f}{b}, \frac{h}{b} \right) - L\left(\frac{\lambda}{b}, \frac{r}{b}, \frac{f_i}{b}, \frac{h_i}{b} \right) + L\left(\frac{\lambda}{b}, \frac{r}{b}, -\frac{f_i}{b}, \frac{h_i}{b} \right) \right] \quad (19)$$

where

$$L\left(\frac{\lambda}{b}, \frac{r}{b}, \frac{f}{b}, \frac{h}{b} \right) = \left\{ \frac{(b - r\lambda) - f(1 - \lambda)}{2(b - r)} \ln \left(\frac{h^2 + (f - b)^2}{h^2 + (f - r)^2} \right) - \left(\frac{1 - \lambda}{b - r} \right) \left[(b - r) + h \tan^{-1} \left(\frac{f - r}{h} \right) - h \tan^{-1} \left(\frac{f - r}{h} \right) \right] \right\} \quad (20)$$

and

$$f_i = \frac{fr^2}{f^2 + h^2}, \quad h_i = \frac{hr^2}{f^2 + h^2} \quad (21)$$

Hence, once the location of the vortex is determined and the position of the image vortices, f_i and h_i , is determined from Equation (21), the interference factor can be computed from Equations (19) and (20). The lift-loss due to interference can then be calculated.

The total lift based on an arbitrary reference area is obtained by adding the components

$$C_L = C_{L_B} + C_{L_W} + C_{L_T} + C_{L_i} \quad (22)$$

where C_{L_i} , a lift-loss, will be a negative quantity. The other three components include both linear and nonlinear contributions.

DRAG CHARACTERISTICS

The total aerodynamic drag acting on a missile is the sum of the zero-lift drag, the induced drag due to angle of attack and/or control surface deflection, and the base pressure drag. It is well-known that the selection of the proper technique for computing the zero-lift drag is determined by the operating speed of the missile. Hence, the methods employed to compute the zero-lift drag of a missile are described for three speed regimes. These speed regimes and their associated limits are defined as follows:

1. Subsonic -- $M < 0.8$
2. Transonic -- $0.8 \leq M \leq 1.2$
3. Supersonic -- $M > 1.2$

The description of C_{D_0} calculations for these speed regimes is followed by a description of the method used to compute the induced drag due to angle of attack and/or tail deflection. The last section presents the total drag.

1. Subsonic Region

The zero-lift drag for subsonic speeds can be expressed as

$$C_{D_0} = C_{D_{0W}} + C_{D_{0T}} + C_{D_{0B}} \quad (23)$$

where each of the components is composed of skin-friction drag and pressure-drag. The pressure drag at subsonic speeds is usually small compared to the drag due to skin friction. Since the flow around high speed missiles of the type being considered here is primarily turbulent, the methods employed are developed assuming the existence of fully turbulent boundary layers.

The zero-lift drag of the body based on an arbitrary reference area is obtained from Section 4.2.3.1 of Reference 9 and can be expressed as

$$C_{D_{0B}} = 1.02 C_{fB} \left[1 + \frac{1.5}{(l_B/d_B)^{3/2}} + \frac{7}{(l_B/d_B)^3} \right] \frac{S_s}{S_{REF}} \quad (24)$$

where C_f , the skin-friction coefficient, is determined using Figure 9.

The wing zero-lift drag as presented in Section 4.1.5.1 of Reference 9 is

$$C_{D_{0W}} = 8.0 C_{fW} \left[1 + 2(t/c) + 100(t/c)^4 \right] \frac{S_w}{S_{REF}} \quad (25)$$

The factor of 8.0 is included to account for the total wetted area of the surfaces and for the existence of four wings. The tail zero-lift drag, $C_{D_{0T}}$, can be obtained by using the tail thickness-to-chord ratio and the total tail area, S_T , in Equation (25).

These three components, based on the same arbitrary reference area, are added as indicated by Equation (23) to give the total subsonic zero-lift drag.

2. Transonic Region

The body drag in the transonic speed regime is composed of compressible skin-friction drag, subsonic pressure drag, and transonic wave drag and is obtained using the techniques presented in Section 4.2.3.1 of Reference 9. The compressible skin-friction drag is obtained using the following equation:

$$C_{D_{f_B}} = 1.02 C_{f_c} \frac{S_s}{S_{REF}} \quad (26)$$

where C_f is a function of both Reynolds number and Mach number and can be determined from Figures 9 and 10. The subsonic pressure drag as extracted from Equation (24) is

$$C_{D_{P_B}} = 1.02 C_{f_B} \left[\frac{1.5}{(l_B/d_B)^{3/2}} + \frac{7}{(l_B/d_B)^3} \right] \frac{S_s}{S_{REF}} \quad (27)$$

The transonic pressure drag is computed using Equation (27) up to a Mach number of 1.0 and then it is decreased linearly from its value at 1.0 to zero at $M=1.2$.

The transonic wave drag of the body is obtained from Figure 11 which presents the wave drag as a function of the nose fineness ratio and Mach number. This figure was constructed from experimental data presented in Reference 10 and by using the curves of wave drag for bodies of revolution as shown in Reference 11 and reproduced here in Figure 12.

The total transonic body zero-lift drag is obtained from the following expression:

$$C_{D_{o_B}} = C_{D_{f_B}} + C_{D_{P_B}} + C_{D_{V_B}} \frac{S_N}{S_{REF}} \quad (28)$$

Experimental results show little increase in the viscous drag of the aerodynamic surfaces from the subsonic to the transonic regime and therefore, the skin-friction drag for the subsonic region is also used in the transonic regime. It may be expressed as follows:

$$C_{D_{O_W}} = 8. C_{f_W} \left[1 + 2 \left(\frac{t}{c} \right) \right] \quad (29)$$

To this wing transonic skin-friction drag is added a drag increment, $\Delta C_{D_{O_W}}$,

which is the transonic wave drag of the wing surfaces. Figure 13 expresses this component as a function of thickness-to-chord ratio, $\frac{t}{c}$, aspect-ratio, and Mach number for rectangular surfaces. For surfaces having swept leading edges, $\Delta C_{D_{O_W}}$ is obtained as for rectangular surfaces and then adjusted to account for the sweep angle using the following equation:

$$\Delta C_{D_{O_W}} = \Delta C'_{D_{O_W}} \left[\cos \Lambda_c/4 \right]^{2.5} \quad (30)$$

The Mach number used in Figure 13 to obtain $\Delta C'_{D_{O_W}}$ for swept lifting surfaces is

$$M' = M \left[\cos \Lambda_c/4 \right]^{1/2} \quad (31)$$

This component of the drag due to the aerodynamic surfaces must be computed for both the wings and tails. The tail contribution is determined in the same manner as the wing contribution above using tail parameters.

The total transonic zero-lift drag is the sum of these components as shown below:

$$C_{D_0} = \left(C_{D_{0W}} + \Delta C_{D_{0W}} \right) \frac{S_W'}{S_{REF}} + \left(\Delta C_{D_{0T}} + C_{D_{0T}} \right) \frac{S_T'}{S_{REF}} + C_{D_{0B}} \quad (32)$$

3. Supersonic Region

A simple empirical method for computing the zero-lift drag for the supersonic speed regime has been developed by assuming a parabolic variation of C_{D_0} with Mach number between 1.2 and 3.0. The resulting equation which is used to compute the zero-lift drag of a missile for Mach numbers greater than 1.2 is

$$C_{D_0} = \left[\frac{C_{D_0}'' - C_{D_0}'}{\sqrt{3} - \sqrt{1.2}} \right] \sqrt{M} + \left[\frac{C_{D_0}'' - C_{D_0}'}{1 - \sqrt{3}/\sqrt{1.2}} \right] + C_{D_0}' \quad (33)$$

where C_{D_0}' and C_{D_0}'' are the values of the total zero-lift drag at Mach numbers 1.2 and 3.0 respectively. It should be noted that although C_{D_0} for Mach numbers greater than 3.0 can be determined from Equation (33), existing hypersonic flow theories would probably provide a more accurate estimate of the zero lift drag.

In order to utilize Equation (33) for determining the variation of C_{D_0} with Mach number, C_{D_0}' and C_{D_0}'' must be specified. C_{D_0}' is determined by using the techniques described in the previous section for the transonic flow regime. Since the magnitude of the supersonic wave drag is heavily dependent on the nose shape of the missile, C_{D_0}'' is determined using one of two methods; the selection of the proper method depends on the missile forebody shape.

In the first method, which is for blunted ogives, pointed ogives, and blunted cones, the supersonic zero-lift drag is a function of Mach number and nose fineness ratio. Since C_{D_0}'' is the zero-lift drag at $M=3.0$, it remains to define its variation with nose fineness ratio. Hoerner, Reference 12, indicates the zero-lift drag for body-fin configurations with slender (high fineness ratio) nose shapes generally peaks at a Mach number of 1.0 to 1.2 and then decreases to approximately its subsonic value plus the transonic wave drag of the lifting surfaces at $M=3.0$. Similar configurations with blunted nose shapes of low fineness ratio reach a peak at about the same Mach number, but decrease very little as the Mach number is increased. Using these trends as an indication of the effect of fineness ratio on the variation of zero-lift drag with Mach number for the supersonic speed regime, C_{D_0}'' for the aforementioned nose shapes is specified as follows:

a) For $(1/d)_N \leq 0.5$, $C_{D_0}'' = C_{D_0}'$

b) For $(1/d)_N \geq 8.0$, $C_{D_0}'' = C_{D_0}' / M=0.8 + \left(\Delta C_{D_0_W} \frac{S_W}{S_{REF}} + \Delta C_{D_0_T} \frac{S_T}{S_{REF}} \right)$

where $C_{D_0}' / M=0.8$ is determined utilizing subsonic flow theory and $\Delta C_{D_0_W}$

and $\Delta C_{D_0_T}$ are the transonic wing and tail wave drag. The variation of the forebody wave drag as a function of Mach number is presented in Figure 12 and was used to construct Figure 14 which is utilized with the following equation:

$$C_{D_0}'' = K_1 C_{D_0}' \quad (34)$$

to compute C_{D_0}'' for nose fineness ratios between 0.5 and 8.0.

In the second method, which is for pointed conical noses, C_{D_o}'' is determined from the following equation:

$$C_{D_o}'' = C_{D_o} / M=0.8 + \left(\Delta C_{D_o} \frac{S_W}{S_{REF}} + \Delta C_{D_o} \frac{S_T}{S_{REF}} \right) + C_{D_o}''_{V_N} \quad (35)$$

where the first two terms are obtained in the same manner described above. The forebody wave drag, which for this type of nose is a function of Mach number and the cone semivertex angle, is obtained from Figure 15.

Once C_{D_o}'' is determined, Equation (30) can be used to compute the zero-lift drag at any Mach number between 1.2 and 3.0. The complete zero-lift drag curve of a missile can now be determined.

4. Induced Drag

The induced drag due to angle of attack and/or tail surface deflection is composed of four drag increments as shown below:

$$C_{D_i} = \Delta C_{D_{B-\alpha}} + \Delta C_{D_{W-\alpha}} + \Delta C_{D_{T-\alpha}} + \Delta C_{D_{T-\delta}} \quad (36)$$

The drag increment for the body is obtained using a method presented in Reference 3, while the induced drag due to the wing and tail is obtained from the drag of an equivalent flat plate normal to the flow. The induced drag on a body of revolution at angle of attack can be expressed as follows:

$$\Delta C_{D_{B-\alpha}} = (k_2 - k_1) \left(\frac{S_b}{S_{REF}} \right) \sin^2 \alpha + n C_{dc} \left(\frac{S_p}{S_{REF}} \right) \sin^3 \alpha \quad (37)$$

The wing induced drag based on an arbitrary reference area is obtained from

$$\Delta C_{D_{W-\alpha}} = C_{D_{FP}} \left(\frac{S_W \sin \alpha}{S_{REF}} \right) \quad (38)$$

where $C_{D_{FP}}$ is the drag of a flat plate normal to the flow field, and is determined from Figure 16. The drag curve shown in Figure 16 has been constructed using the three-dimensional subsonic drag coefficient for a flat plate normal to the flow (Reference 13) and the variation of the drag coefficient with Mach number for the two-dimensional flat plate (Reference 12). The equivalent flat plate area of the wing is taken as the projection of the wing area on the normal plane, $S_W \sin \alpha$. Similarly, the drag increment of the tail surface at angle of attack is

$$\Delta C_{D_{T-\alpha}} = C_{D_{FP}} \left(\frac{S_T \sin \alpha}{S_{REF}} \right) \quad (39)$$

The drag increment due to deflection of the tail surface cannot be obtained directly since the drag increment is not a linear function of local angle of attack. The total drag increment for the tail surface can be expressed as

$$\Delta C_{D_{T-\alpha}} + \Delta C_{D_{T-\delta}} = C_{D_{FP}} \left[\frac{S_T \sin(\alpha + \delta)}{S_{REF}} \right] \quad (40)$$

where $(\alpha + \delta)$ is the local tail angle of attack. It now becomes a simple matter to obtain the drag increment due to the deflection of the control surface by taking the difference between Equations (39) and (40). Thus

$$\Delta C_{D_{T-\delta}} = \left(\Delta C_{D_{T-\alpha}} + \Delta C_{D_{T-\delta}} \right) - \Delta C_{D_{T-\alpha}} \quad (41)$$

6. Total Drag

The total drag can be expressed as follows:

$$C_D = C_{D_o} + C_{D_i} \quad (42)$$

where C_{D_i} is determined using Equation (36) and C_{D_o} is obtained from

Equation (23), (32), or (33) depending on the speed of the missile.
Base pressure drag is not included in Equation (42).

PITCHING MOMENT CHARACTERISTICS

The total pitching moment acting on the missile is the sum of the moments due to the lift and drag forces acting on the body, wings, and tails. Most methods for computing the pitching moment (References 1, 6, and 9) consider only the moment due to lift. This is valid only for small angles of attack. If large angles of attack are to be considered, the moment must include the drag contribution. In general, the body longitudinal pitching moment is determined directly; while the other components are determined only after the centers of pressure of the wing and tail surfaces are specified.

The body-alone pitching moment about its center of gravity is obtained from the method of Allen, References 2 and 3. The expression is

$$\begin{aligned}
 C_{m_B} &= (k_2 - k_1) \left(\frac{V_B - S_B \left(l_B - X_{CG} \right)}{S_{REF} l_{REF}} \right) \sin 2\alpha \cos \frac{\alpha}{2} \\
 &+ n C_{dc} \left(\frac{S_P}{S_{REF}} \right) \left(\frac{X_{CG} - X_P}{l_{REF}} \right) \sin^2 \alpha
 \end{aligned} \tag{43}$$

where $(k_2 - k_1)$, C_{dc} , and n are determined from Figures 3 and 4. This moment coefficient, C_{m_B} , is based on an arbitrary reference length and area.

As noted above, the center of pressure locations for the wing and tail surfaces must be specified before their pitching moments can be determined. It must be remembered that the linear lift of the aerodynamic surfaces is composed of two components: the lift on the surface in the presence of the missile body and the additional lift on the body due to the presence of a lifting surface. This means that in order to determine the linear pitching moment caused by the lifting surfaces, it is necessary to specify the center of pressure location for each of these linear components.

The center of pressure of the lift on the wing in the presence of the body as measured from the junction of the wing leading edge and the

body is obtained using Figure 17. The reference point is transferred to the nose by using the following equation

$$X_{WB} = \left(\frac{\bar{X}}{C_r} \right)_{WB} (C_r)_W + l_W \quad (44)$$

where $\left(\frac{\bar{X}}{C_r} \right)_{WB}$ is obtained from Figure 17a for subsonic speeds and from Figure 17b for supersonic speeds.

The center of pressure of the additional lift on the body in the presence of the wing is obtained using Figure 18 if the flow is subsonic, and either Figure 19 or 20 if the flow is supersonic. For the case of supersonic flow, Figure 19 is used if

$$\beta \text{AR} (1 + \lambda) \left(1 + \frac{l}{m\beta} \right) \leq 4.0$$

and Figure 20 is used if the above quantity is greater than 4.0. The center of pressure, \bar{X}_{BW} , as obtained from the aforementioned figures, is referred to the nose by using the following equation.

$$\left(X_{CP} \right)_{BW} = \left(\frac{\bar{X}}{C_r} \right)_{BW} (C_r)_W + l_W \quad (45)$$

$\left(X_{CP} \right)_{BT}$ The location of both centers of pressure for the tails, $\left(X_{CP} \right)_{TB}$ and $\left(X_{CP} \right)_{BT}$, can also be obtained from the above procedure.

The centers of pressure for a given lifting surface are combined to obtain a single average center of pressure location for each set of aerodynamic surfaces. For example, the average center of pressure of the wings is obtained by computing the total pitching moment due to the wings and dividing by the wing normal force. The pitching moment about the

nose of the body due to the wing is

$$C_{mW} = \left[\left(C_{L_{WB}} + C_{L_{WV}} + C_{L_i} \right) \cos \alpha + \Delta C_{D_{W-\alpha}} \sin \alpha \right] \left[\left(x_{CP} \right)_{WB} / l_{REF} \right] + \left[C_{L_{BW}} \cos \alpha \right] \left[\left(x_{CP} \right)_{BW} / l_{REF} \right] \quad (46)$$

where all of the above terms have been previously defined. The average wing center of pressure as measured from the nose of the body can now be defined as

$$\left(x_{CP} \right)_W = \frac{C_{mW} l_{REF}}{\left(C_{L_W} + C_{L_i} \right) \cos \alpha + \Delta C_{D_{W-\alpha}} \sin \alpha} \quad (47)$$

Similarly, the tail pitching moment and center of pressure including the effect of surface deflection can be expressed as

$$C_{mT} = \left[\left(C_{L_{TB-\alpha}} + C_{L_{TV}} + C_{L_i} \right) \cos \alpha + C_{L_{TB-\delta}} + \left(\Delta C_{D_{T-\alpha}} + \Delta C_{D_{T-\delta}} \right) \sin \alpha \right] \left[\left(x_{CP} \right)_{TB} / l_{REF} \right] + \left[C_{L_{BT-\alpha}} \cos \alpha + C_{L_{BT-\delta}} \right] \left[\left(x_{CP} \right)_{BT} / l_{REF} \right] \quad (48)$$

$$\left(x_{CP} \right)_T = \frac{C_{mT} l_{REF}}{\left(C_{L_{T-\alpha}} + C_{L_i} \right) \cos \alpha + C_{L_{T-\delta}} + \left(\Delta C_{D_{T-\alpha}} + \Delta C_{D_{T-\delta}} \right) \sin \alpha} \quad (49)$$

where

$$C_{L_{T-\alpha}} = C_{L_{TB-\alpha}} + C_{L_{BT-\alpha}} + C_{L_{TV}}$$

and

$$C_{L_{T-\delta}} = C_{L_{TB-\delta}} + C_{L_{BT-\delta}}$$

The total longitudinal pitching moment of the missile about the missile's center of gravity may be expressed as

$$C_m = C_{m_B} + C_{m_W} \left[\frac{x_{CG} - (x_{CP})_W}{(x_{CP})_W} \right] + C_{m_T} \left[\frac{x_{CG} - (x_{CP})_T}{(x_{CP})_T} \right] \quad (50)$$

COMPUTER PROGRAM DESCRIPTION

The method presented herein for obtaining the static aerodynamic characteristics of a missile has been programmed for use on the IBM 7090 computer and other compatible digital computers. The program, Table 1, is written in Fortran II and requires only the geometric characteristics of the missile and its flight conditions as inputs. The output consists of the static longitudinal aerodynamic characteristics in coefficient form, the center-of-pressure location for the body, wings, and tails, the lift-curve slope for each set of lifting surfaces, and the components of the lift and normal force coefficients. The force components referred to are the body, wing, and tail lift and normal force coefficients and the coefficient representing the lift-loss due to downwash. The aerodynamic characteristics are output in both the stability and body axis systems (Figure 21). Provision has been made for a third lifting surface to account for the possibility of using strakes in combination with two other sets of lifting surfaces (Figure 1).

The program itself consists of a main program and three subroutines--GEOSUB, CLASUB, and CATSUB. The first subroutine performs some initial geometric computations, determines the nose wave drag constant (Figure 14), and obtains the Reynolds number per foot based on the altitude input to the program. Subroutine CLASUB determines the lift-curve slope of the lifting surfaces from curve fits employed to represent the curves presented in Figure 6. The last subroutine, CATSUB, obtains the body-wing and body-tail interference factors, computes the center of pressure location as a function of the root chord of the lifting surfaces, and determines the crossflow drag coefficient for the lifting surfaces.

The program computes the static force and moment coefficients for typical missile configurations at specified angles of attack, control surface deflection angles and Mach numbers. The angle of attack range is -180° to $+180^\circ$, and any control surface deflection within this range may be used. The Mach number is limited to 3.0 only because the drag prediction methods are valid up to this particular Mach number. The program can be used for configurations at $M > 3.0$; however, the drag predictions above this limit should be used with caution. The computer

program can be used to obtain build-up information; that is, the aerodynamic characteristics of the missile body alone, the body-wing configuration, and the body-tail configuration. This information may be obtained by simply setting the appropriate parameters to zero.

The inputs to the computer program with their FORTRAN symbols are presented in Table 2. The format for preparing the input cards is presented in Table 3. It should be noted that if a configuration does not have a control surface, e.g. a body alone configuration, the number of control surface deflection angles should be set at one and the deflection angle, itself, would be 0.0 degrees. Cards 11 and 12 are used only when the number of angles of attack require their use. The output variables are defined in Table 4. There is no limit to the number of data decks which may be stacked together and run at the same time.

COMPARISON OF THEORY WITH EXPERIMENTAL DATA

Numerous comparisons of theory with experiment have been made in order to establish and verify the accuracy of the method. Figure 22 presents the configurations used for comparison. Configuration 1 is a strake-tail configuration. Figure 23 presents the comparison between the experimental data and the theoretical results obtained from the method described herein. The theoretical lift and drag are within 15 percent of the experimental data for $M = 0.7$. The deviation for $M = 1.1$ does increase to about 25 percent. However, it should be noted that the accuracy of the method to 30 degrees is good. The method also predicts the center of pressure location very satisfactorily.

Configuration 2 is a wing-controlled vehicle. Comparisons are presented in Figure 24 for Mach numbers of 1.12 and 2.16 and for wing deflection angles of 0 and -10 degrees. Although high angle of attack data is not available for this configuration, it appears that the comparatively high theoretical lift shown by Configuration 1 may be more pronounced for Configuration 2. This trend is seen by the decrease in the slope of the experimental lift data at 25 to 30 degrees angle of attack, Figure 24. The comparison with the wing deflected down 10 degrees appeared to be good.

Data for Configuration 3 is presented in Figure 25. Configuration 4 was tested by the Cornell Aeronautical Laboratory to 180 degrees angle of attack, Reference 14. Comparison of theory with experimental data is presented in Figure 26.

CONCLUSIONS

A method for predicting the static, longitudinal aerodynamic characteristics of low aspect-ratio missiles operating at angles of attack to 180 degrees has been developed. The method is valid for a wide speed range and considers control surface deflections. A computer program, written to facilitate use of the method, has been described. Results obtained using the method have been compared with wind tunnel data and acceptable agreement has been demonstrated.

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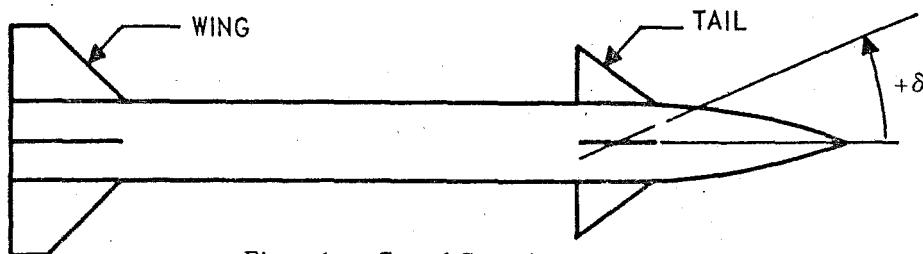


Figure 1a -- Canard Control

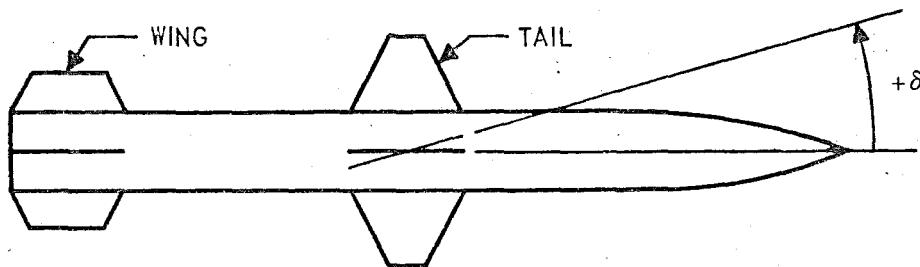


Figure 1b -- Wing Control

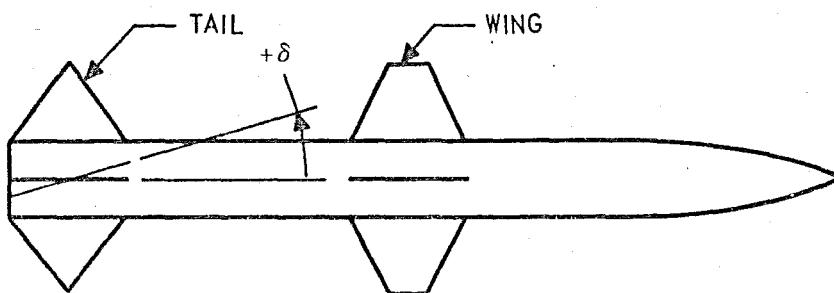


Figure 1c -- Tail Control

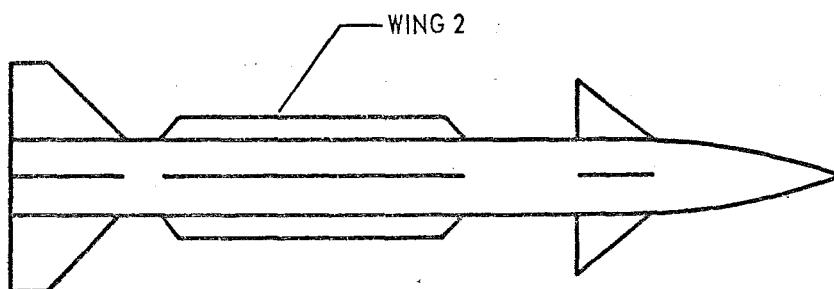


Figure 1d -- Three Surfaces -- Any Mode of Control

Figure 1 -- Typical Missile Configurations

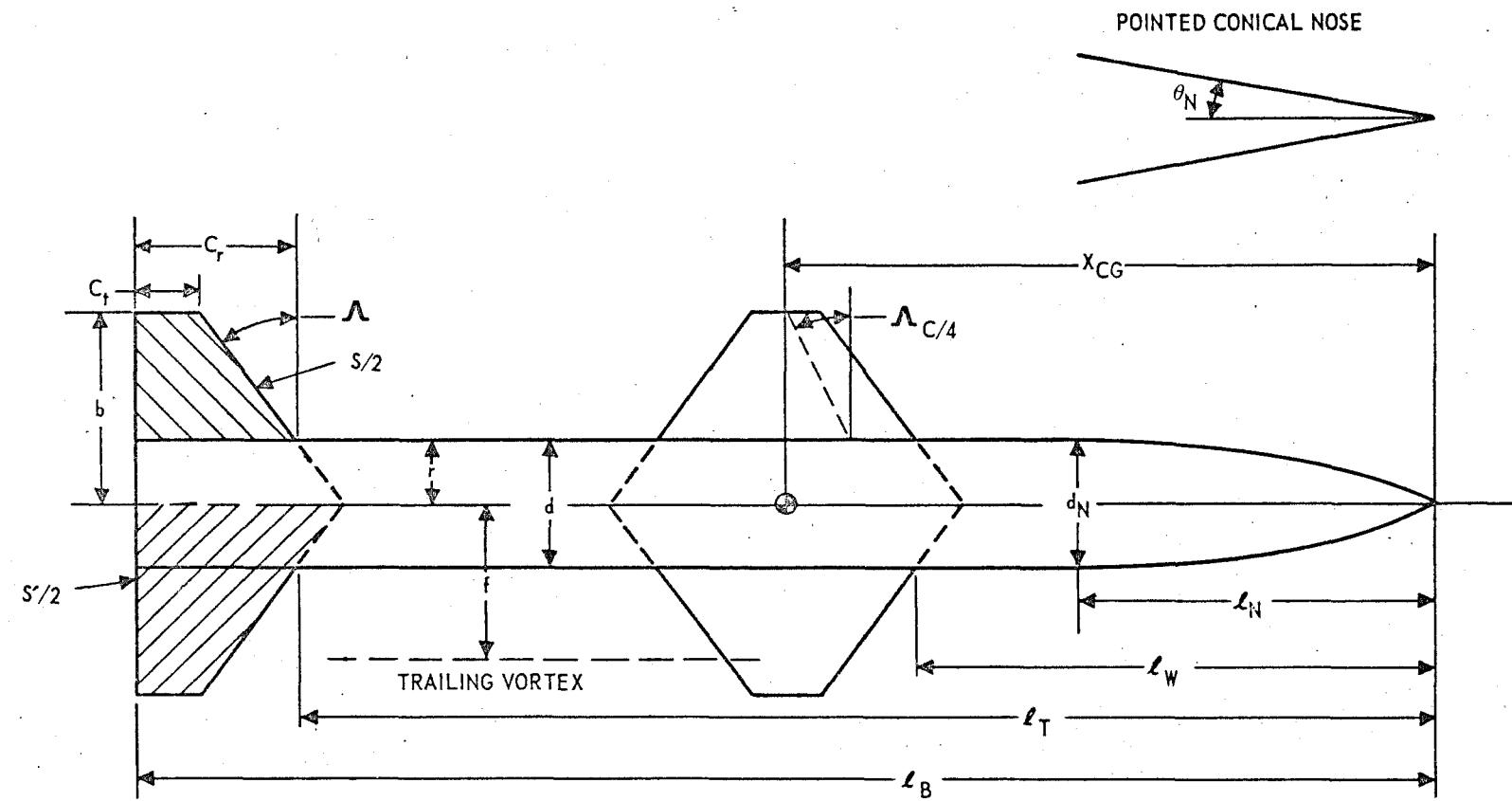


Figure 2 – General Geometric Characteristics

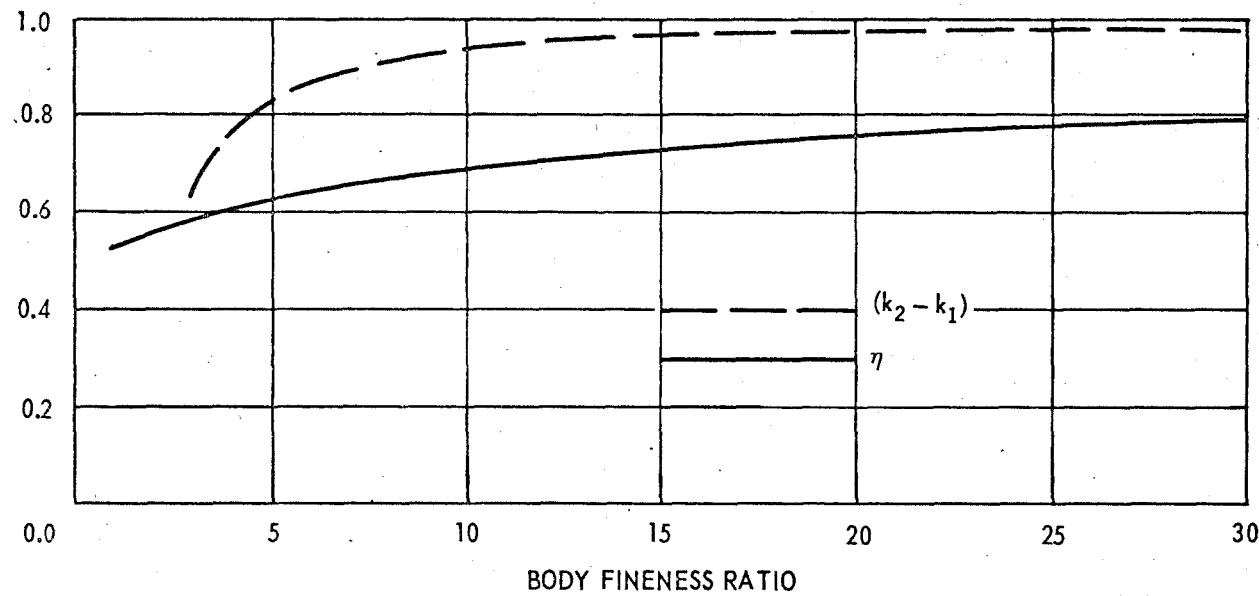


Figure 3 – Parameters Used to Compute Body Normal Force and Pitching Moment (from Reference 3)

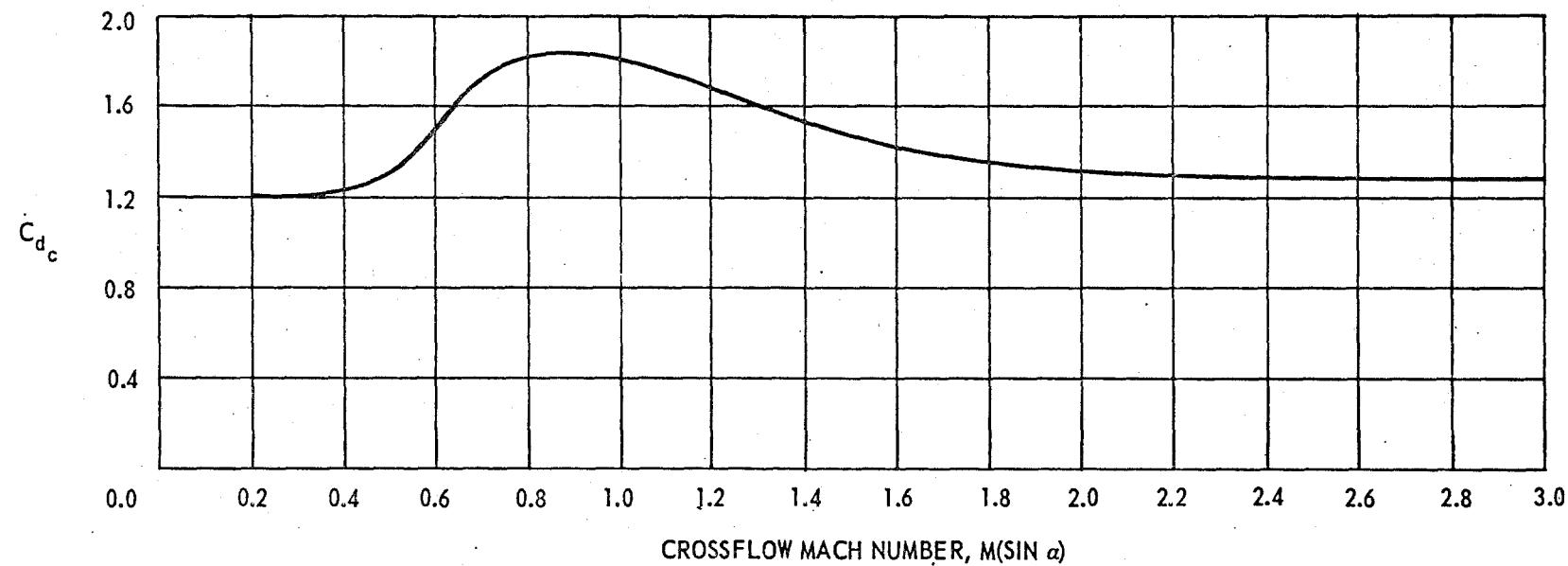


Figure 4 – Crossflow Drag Coefficient as a Function of Mach Number (from Reference 3)

Figure 5 – Linear Lift Interference Factors (from Reference 7)

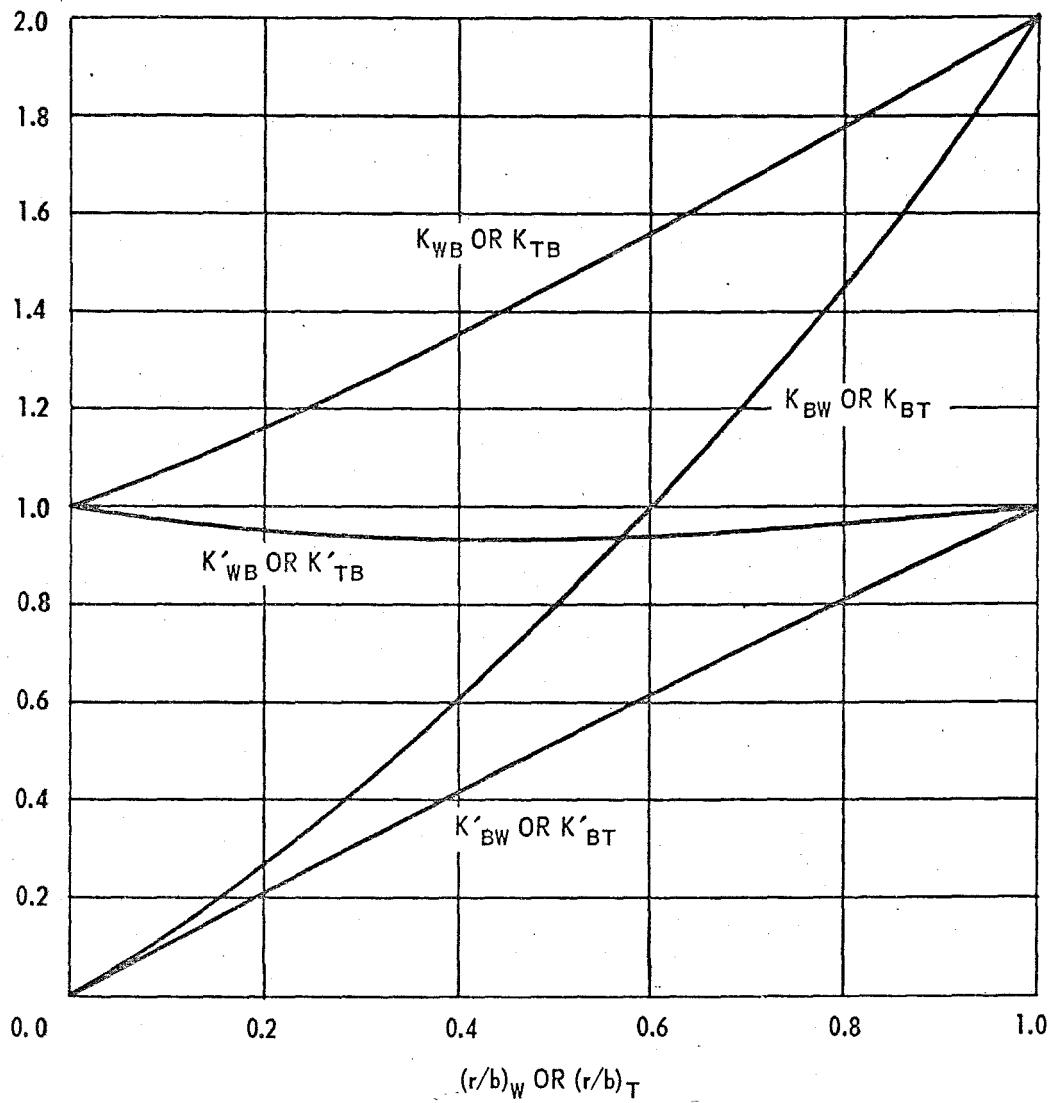


Figure 5a – $(\beta\text{AR})(1 + \lambda)(1/m\beta + 1) \leq 4$

15

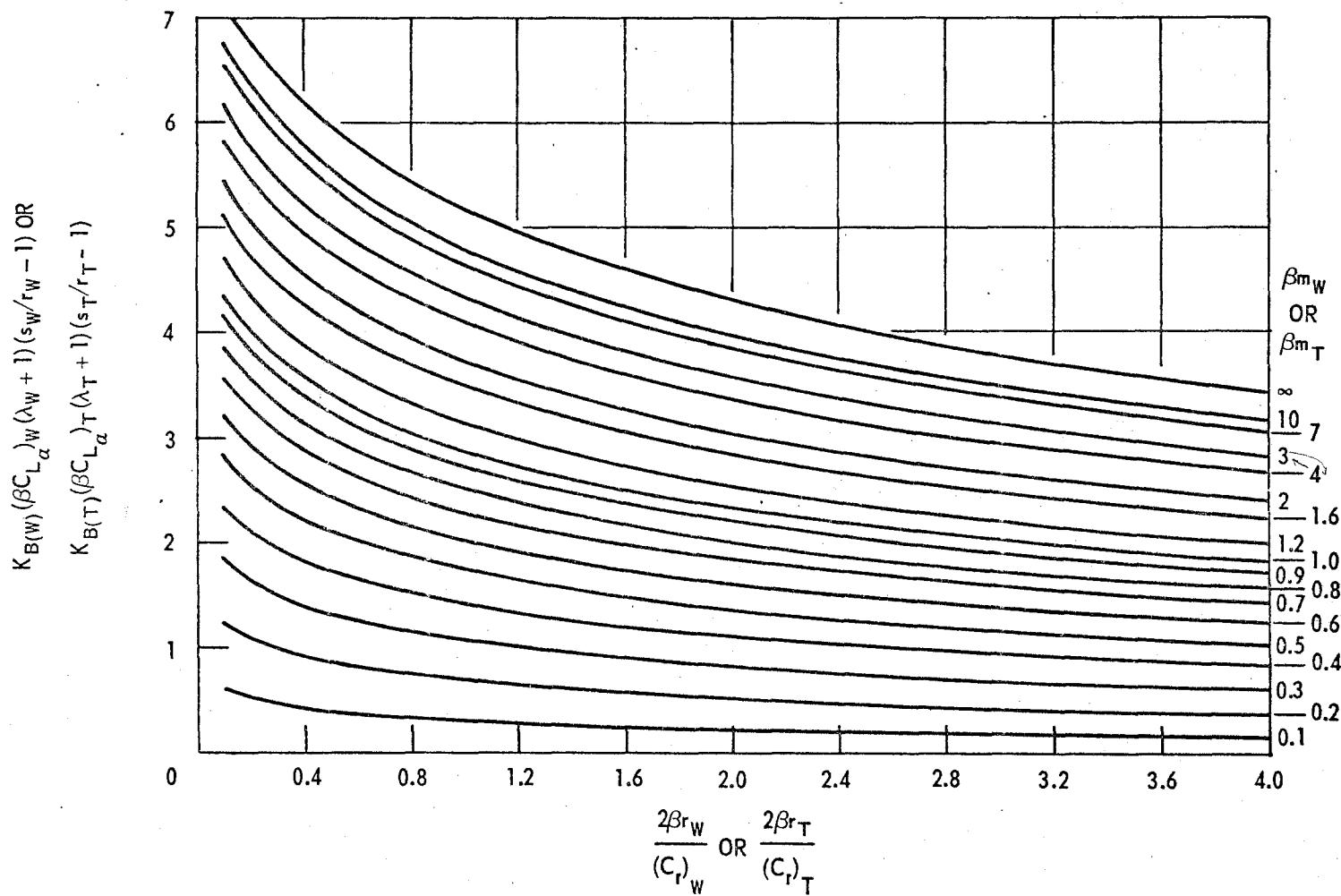


Figure 5b - $(\beta \text{AR})(1 + \lambda)(1/m\beta + 1) > 4$; with Afterbody

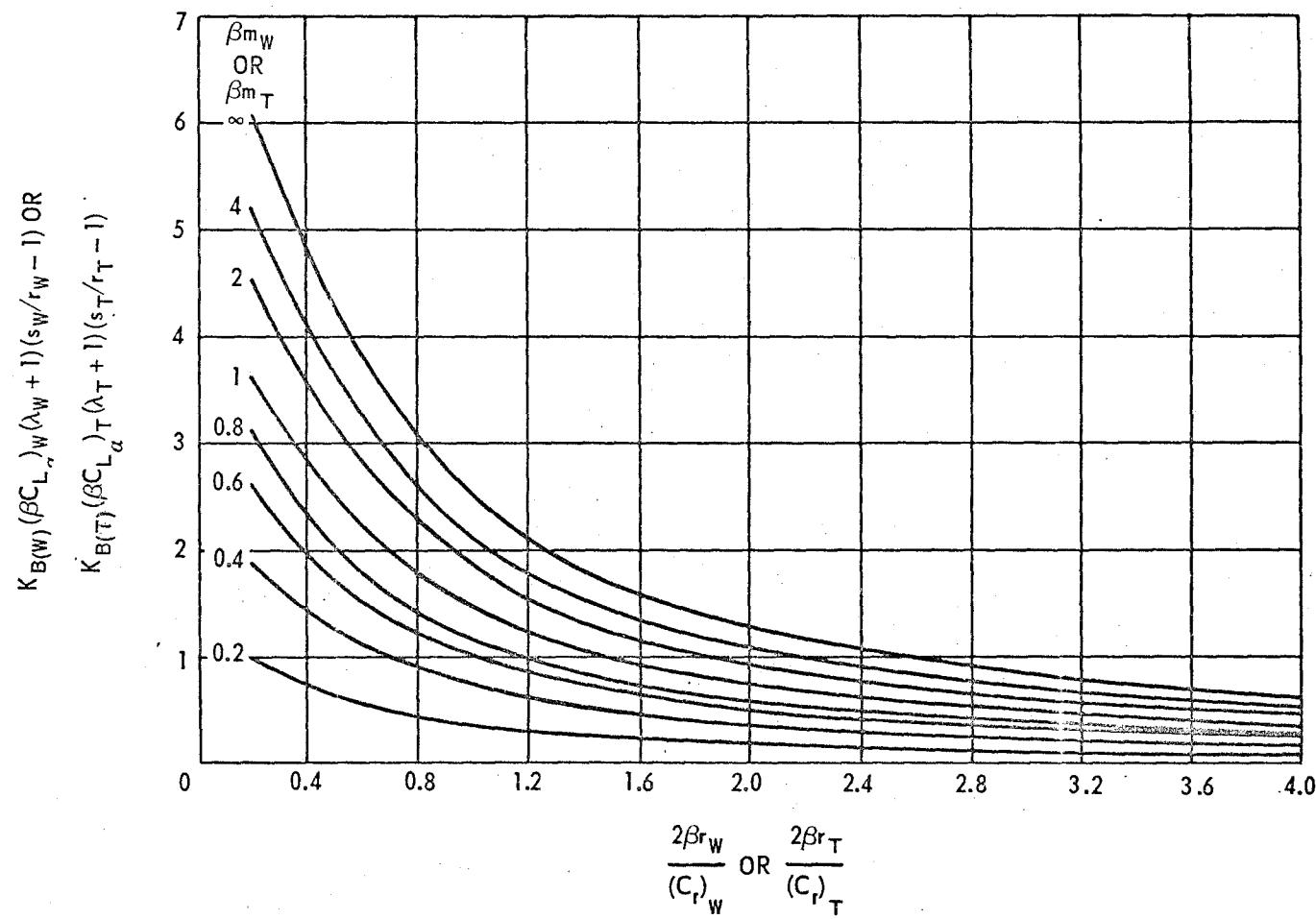


Figure 5c -- $(\beta \text{AR})(1 + \lambda)(1/m\beta + 1) > 4$; No Afterbody

Figure 6 – Lift Curve Slope for Wings and Tails (from Reference 8)

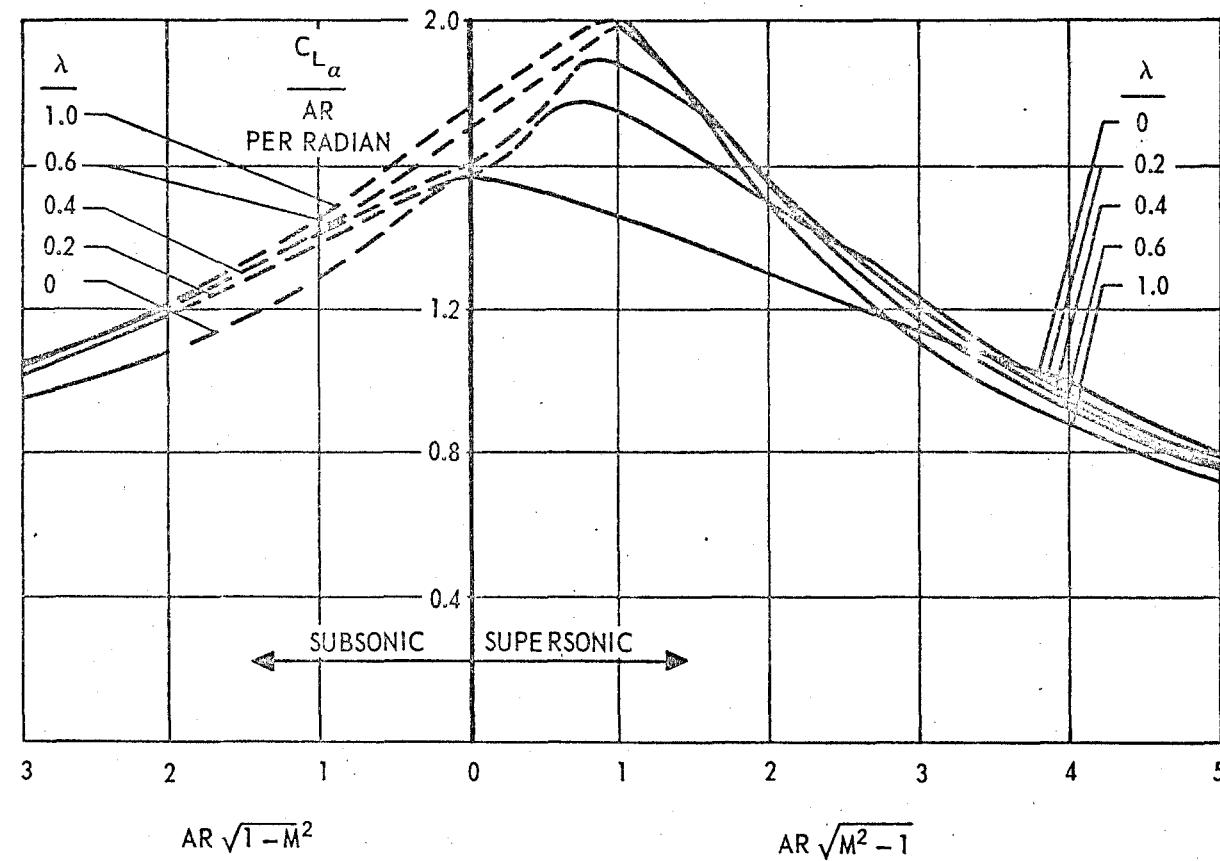


Figure 6a — Unswept Trailing Edge

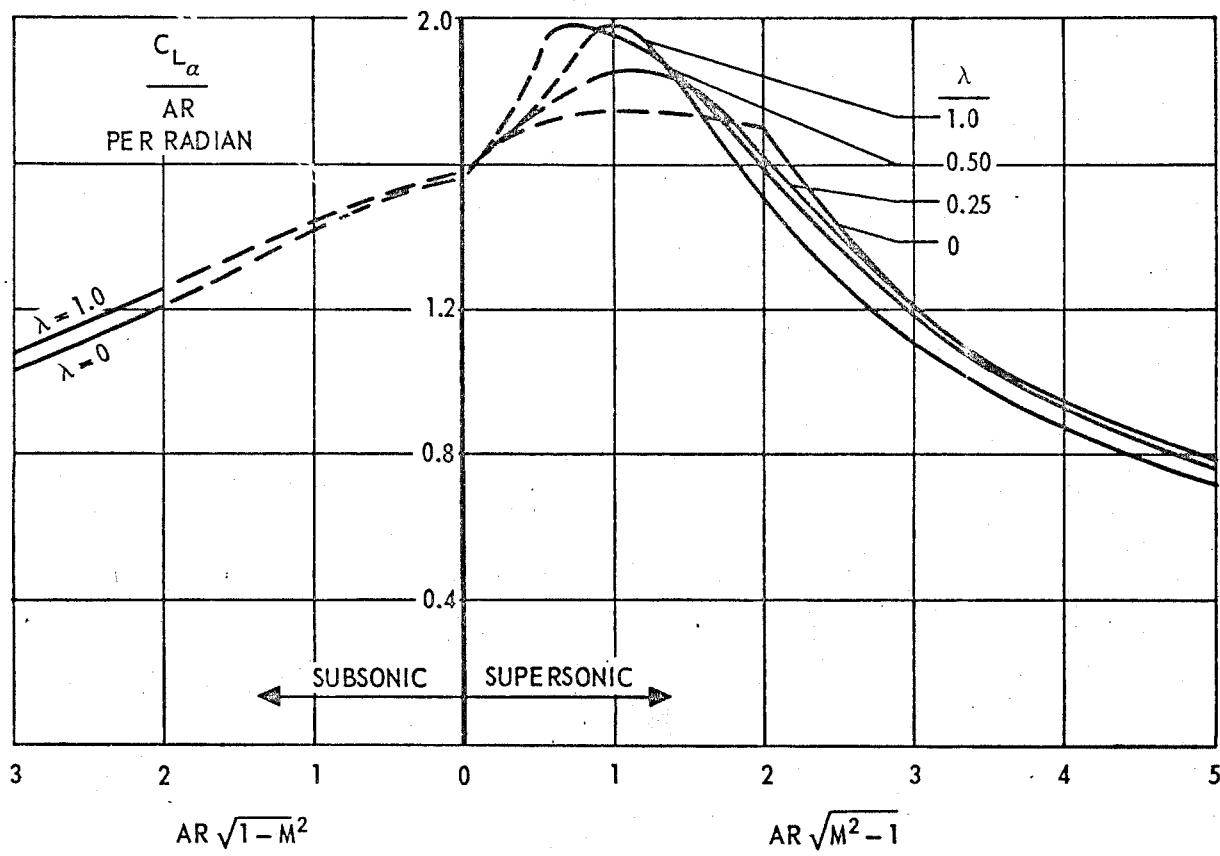


Figure 6b — Unswept Mid Chord

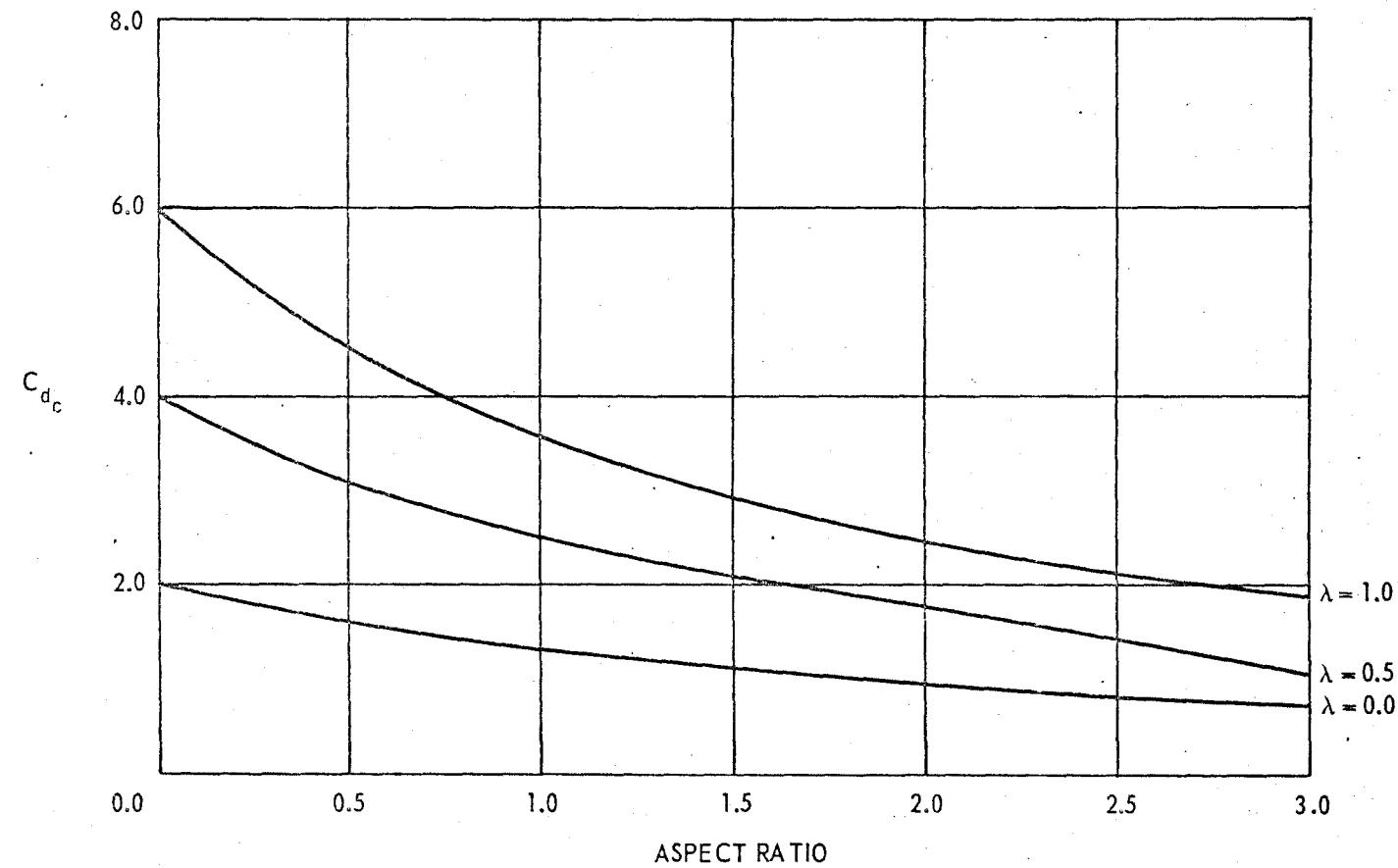


Figure 7 – Crossflow Drag Coefficient for Wings and Tails as a Function of Aspect Ratio and Taper Ratio

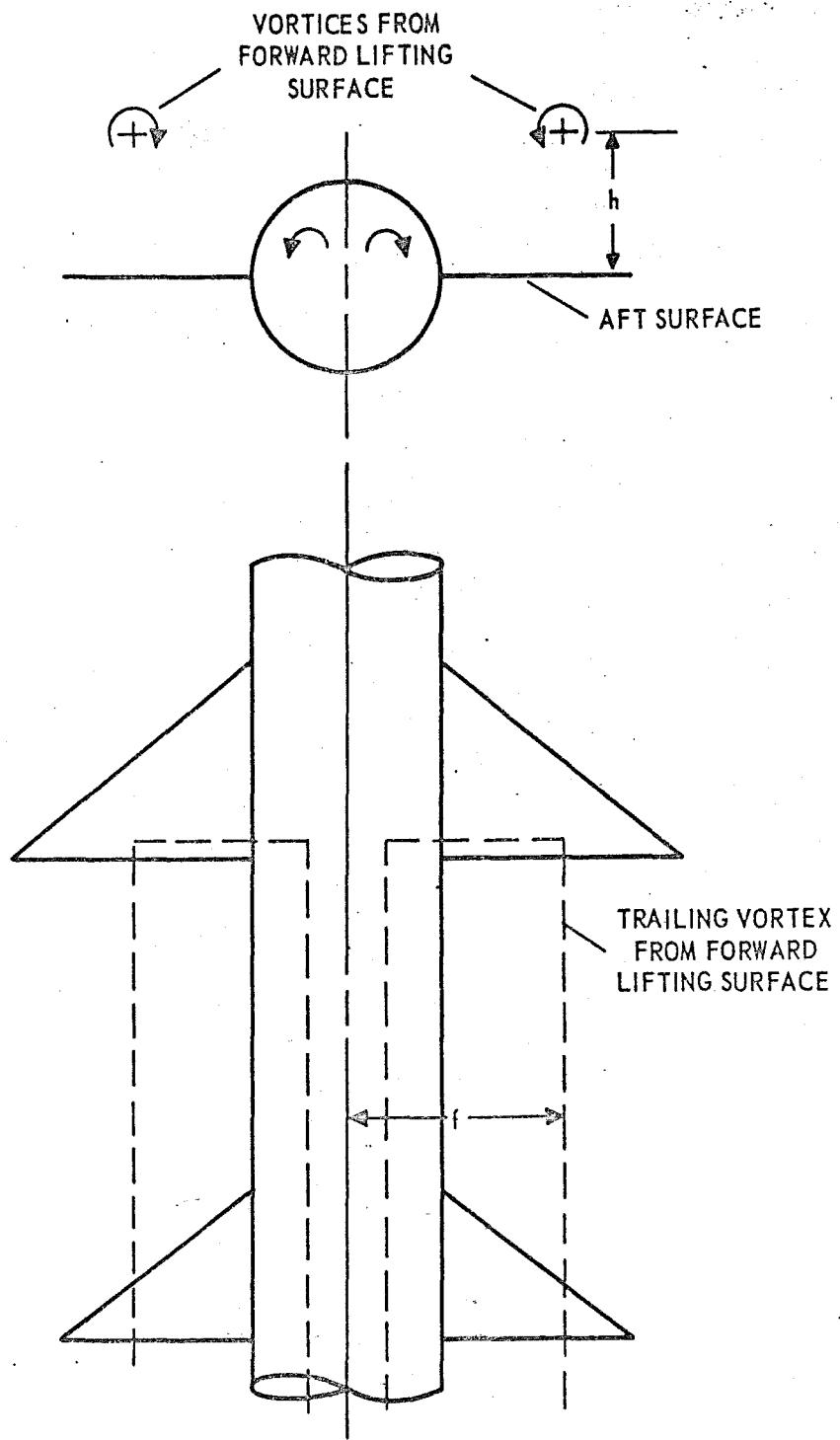


Figure 8 – Vortex Model Used to Determine the Lift Loss due to Downwash (from Reference 1)

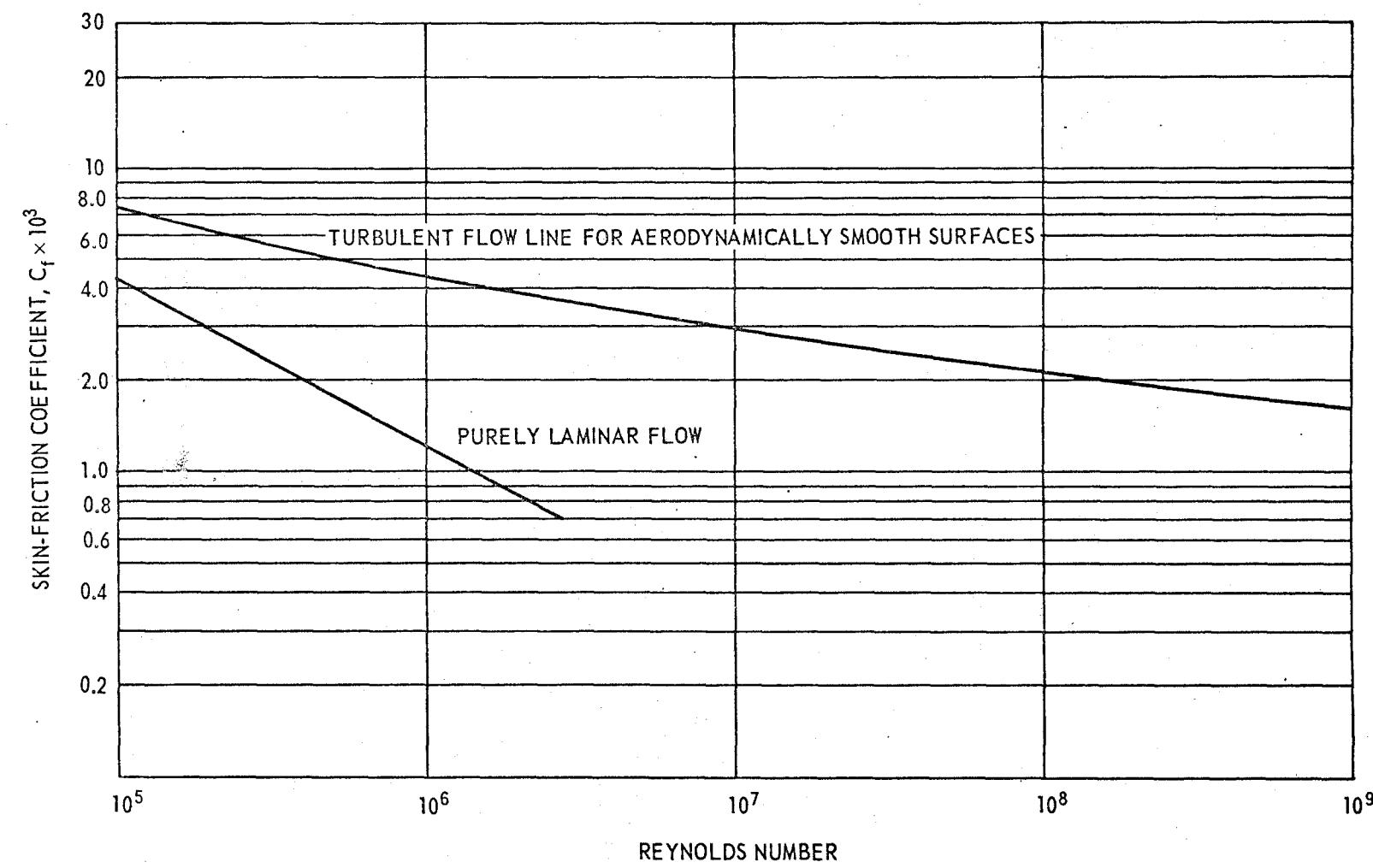


Figure 9 – Incompressible Skin Friction Coefficient (from Reference 9)

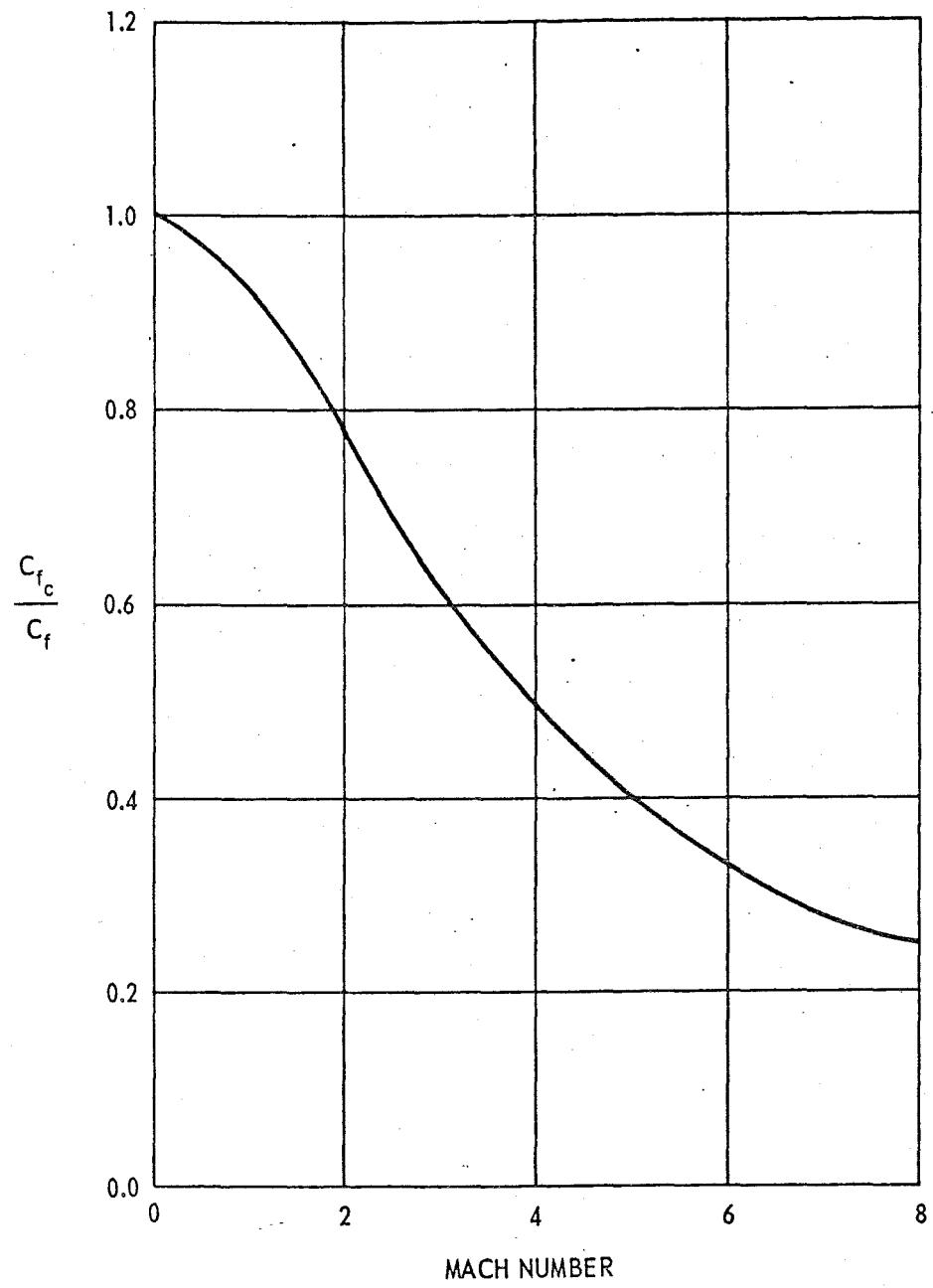


Figure 10 – Compressibility Effect on Turbulent Skin Friction (from Reference 9)

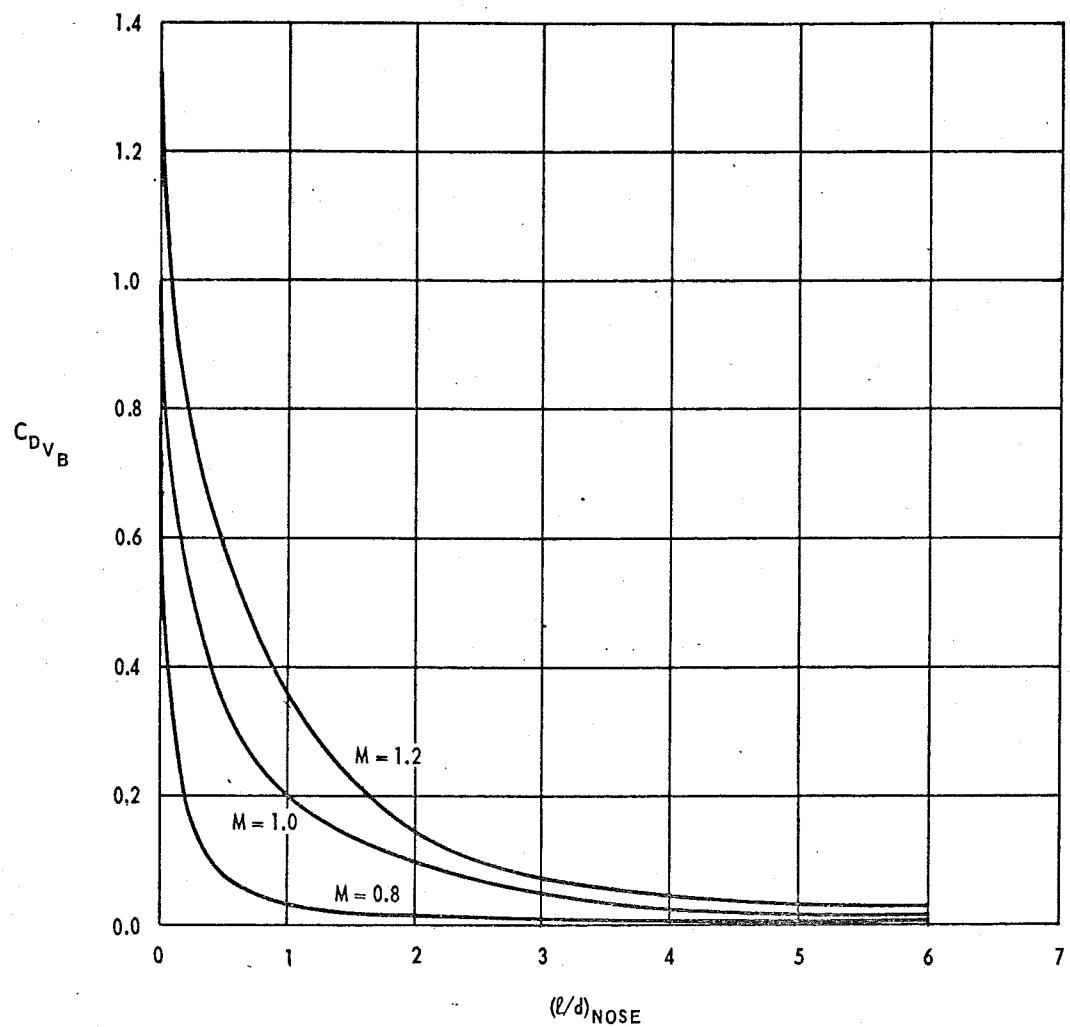


Figure 11 – Transonic Wave Drag for Ogival and Blunted Conical Forebodies

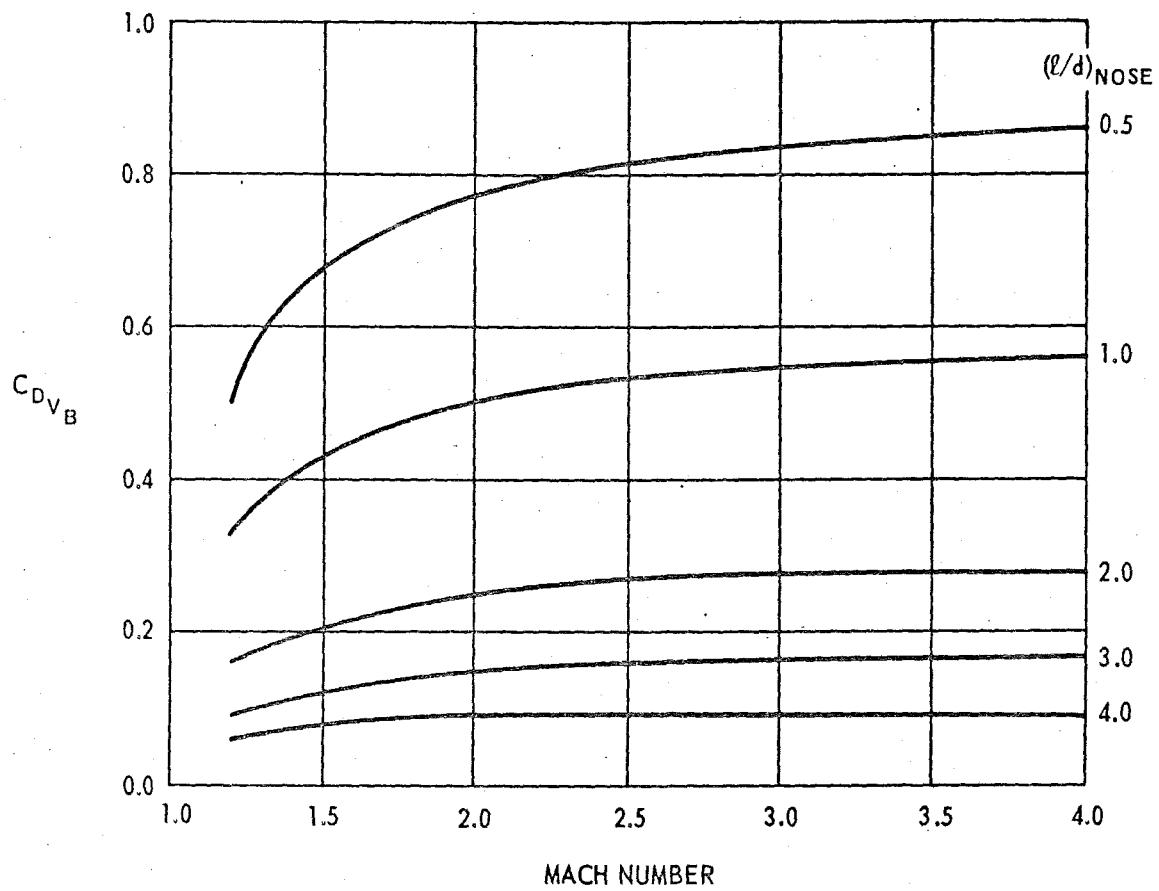


Figure 12 – External Wave Drag of Blunt Forebodies (from Reference 11)

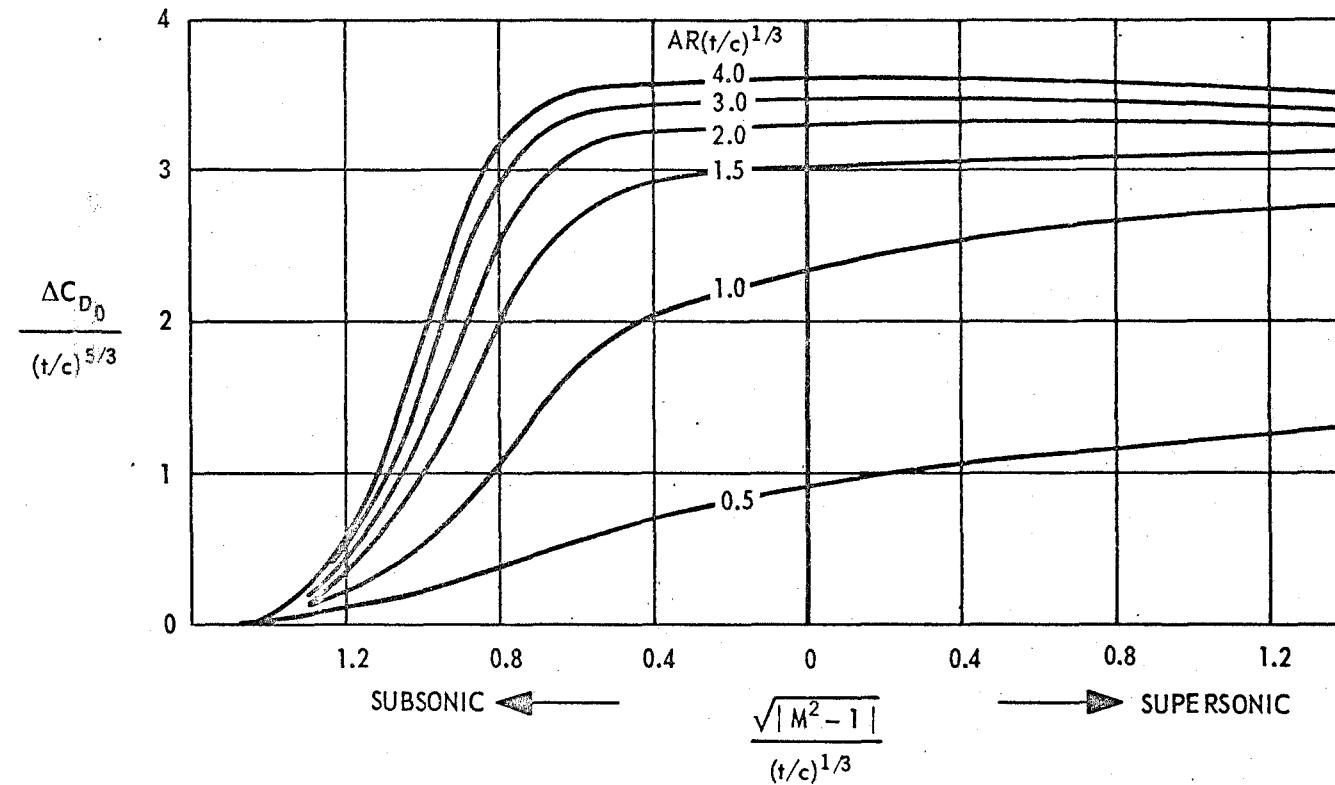


Figure 13:— Transonic Zero-Lift Wing Wave Drag for Unswept Wings (from Reference 9)

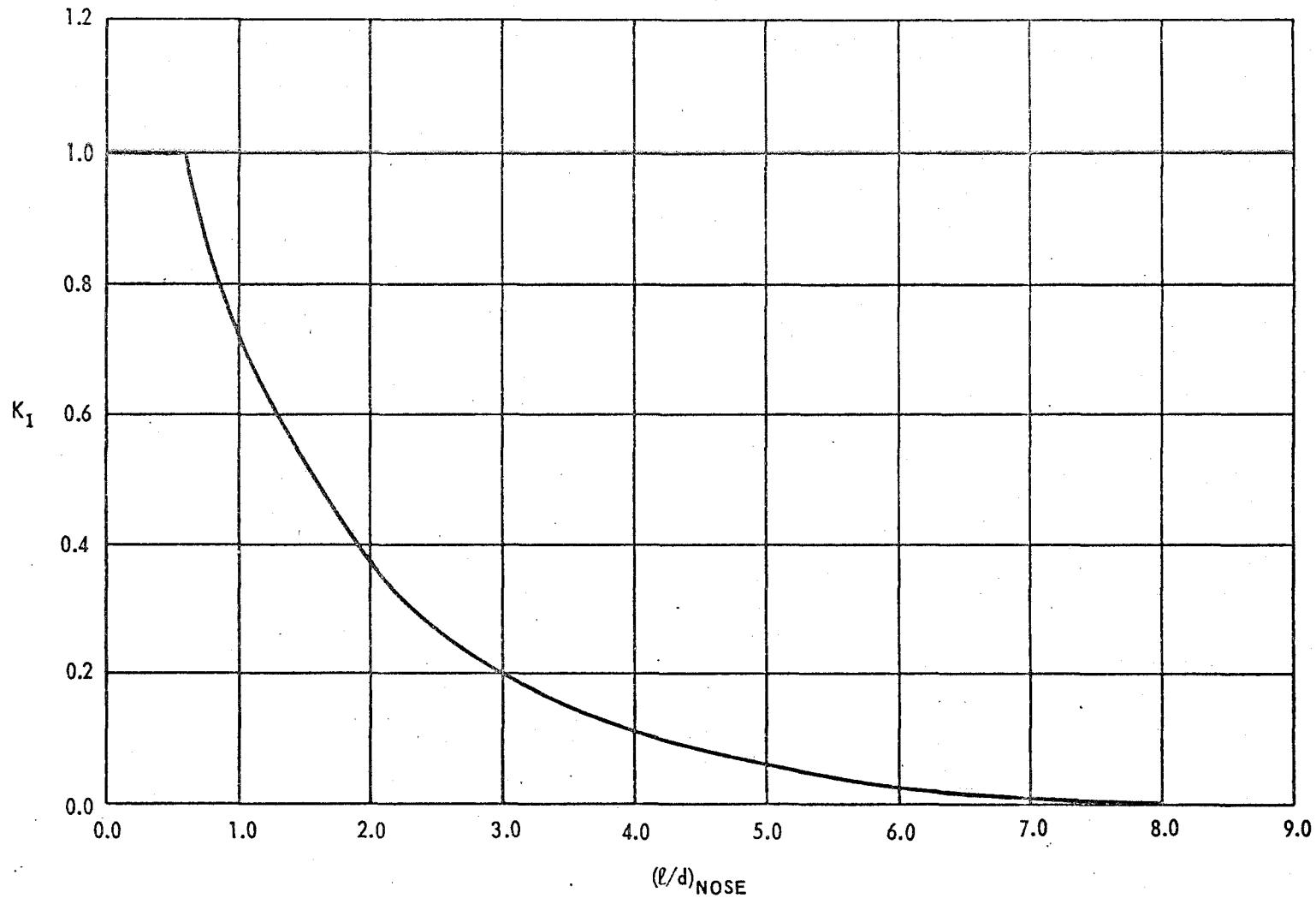


Figure 14 – Ratio of Wave Drag for Noses of Various Fineness Ratios to the Wave Drag for a Hemispherical Nose

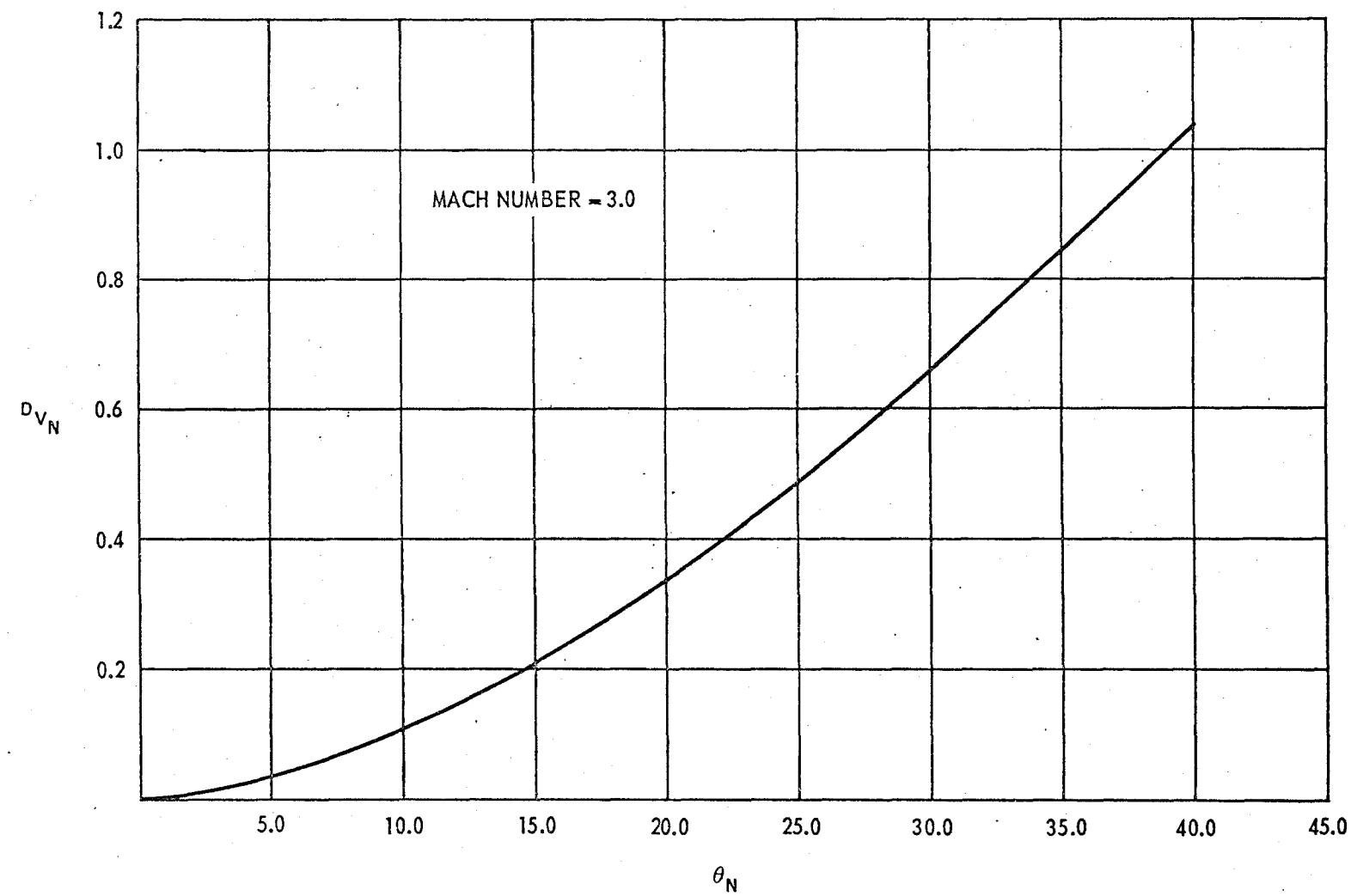


Figure 15 – Wave Drag of a Pointed Conical Nose (from Reference 6)

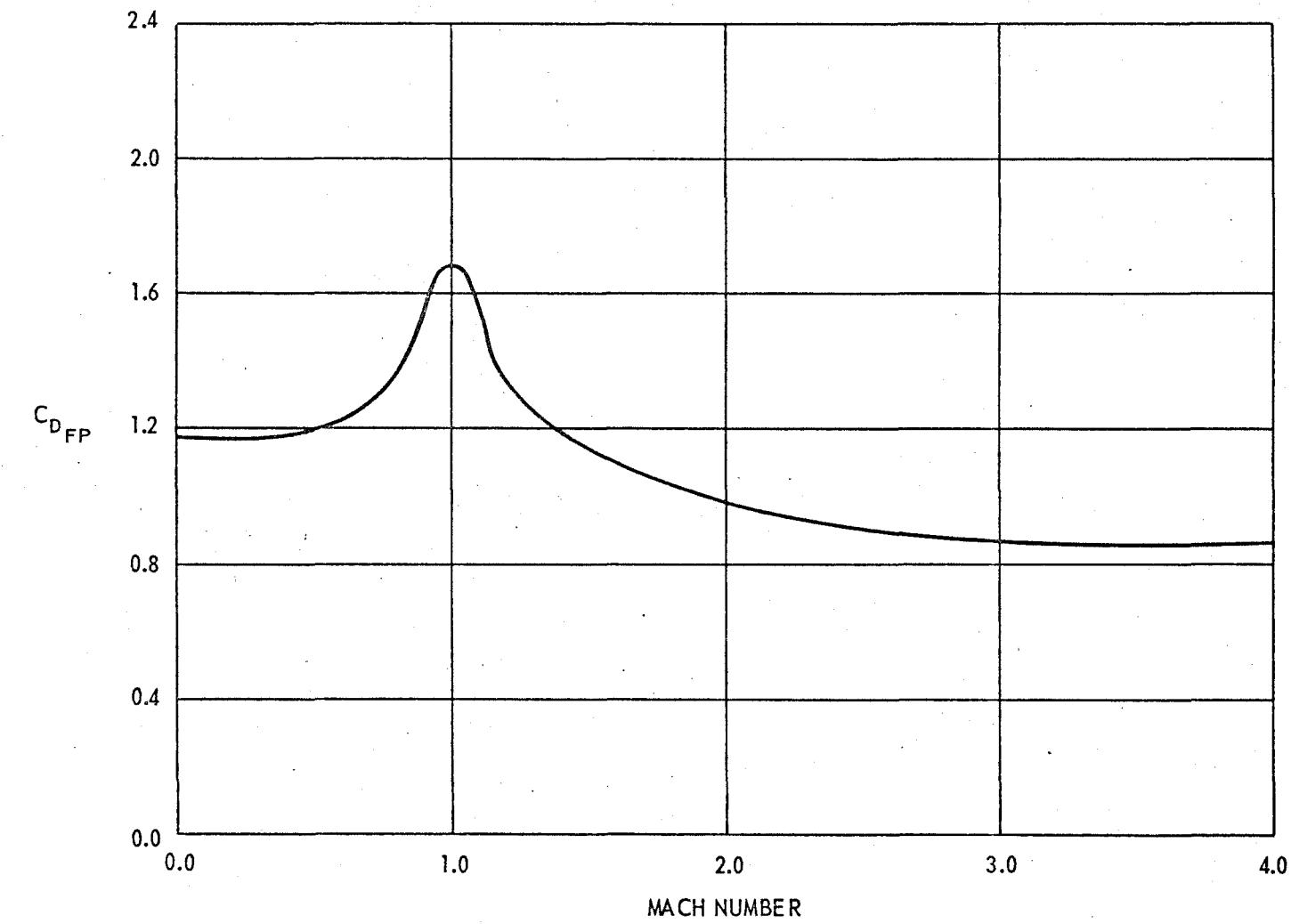


Figure 16 – Drag Coefficient for a Flat Plate Normal to the Flow

Figure 17 – Lifting Surface Center of Pressure as a Function of Effective Aspect Ratio
(from Reference 1)

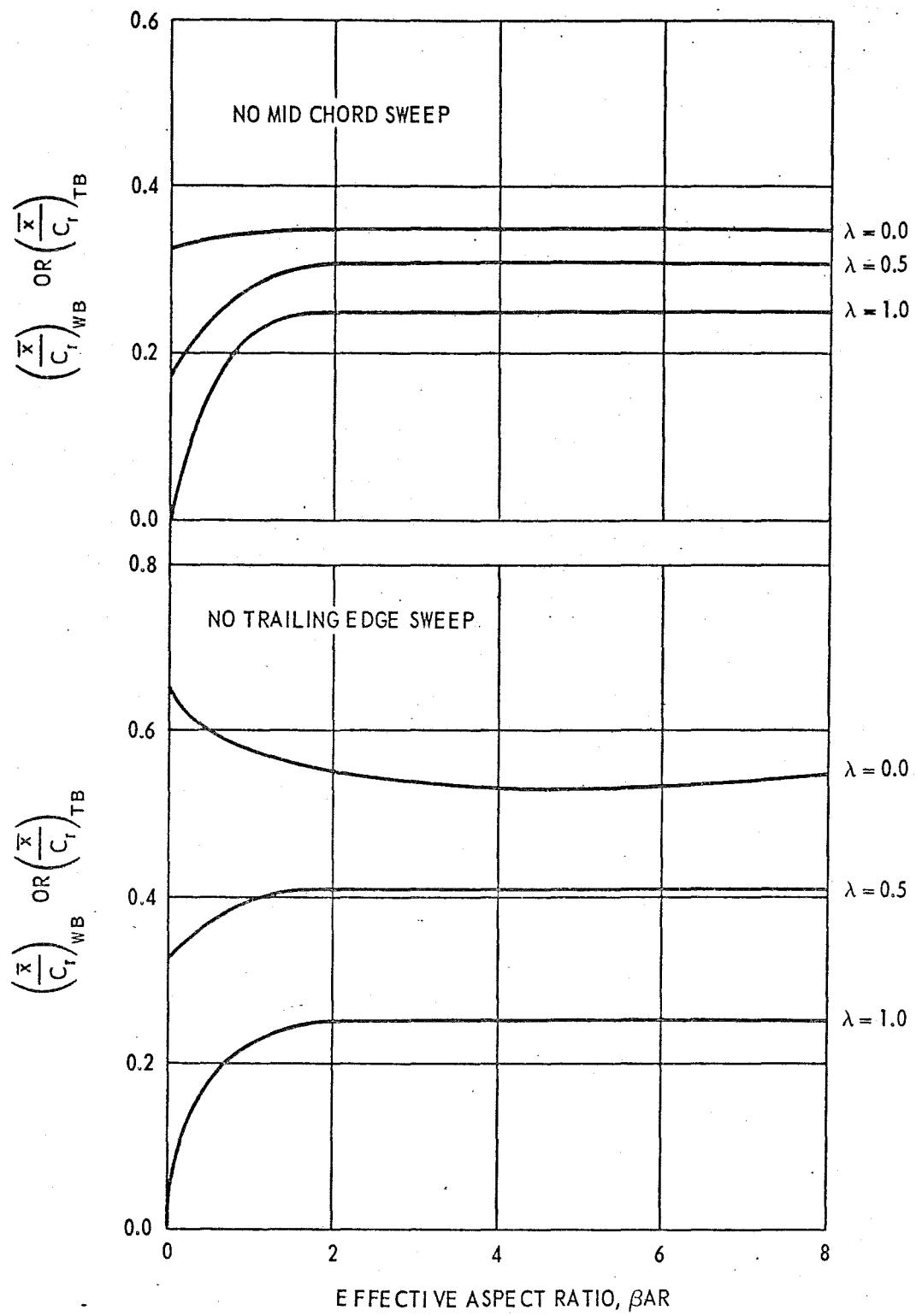


Figure 17a – $M \leq 1.0$

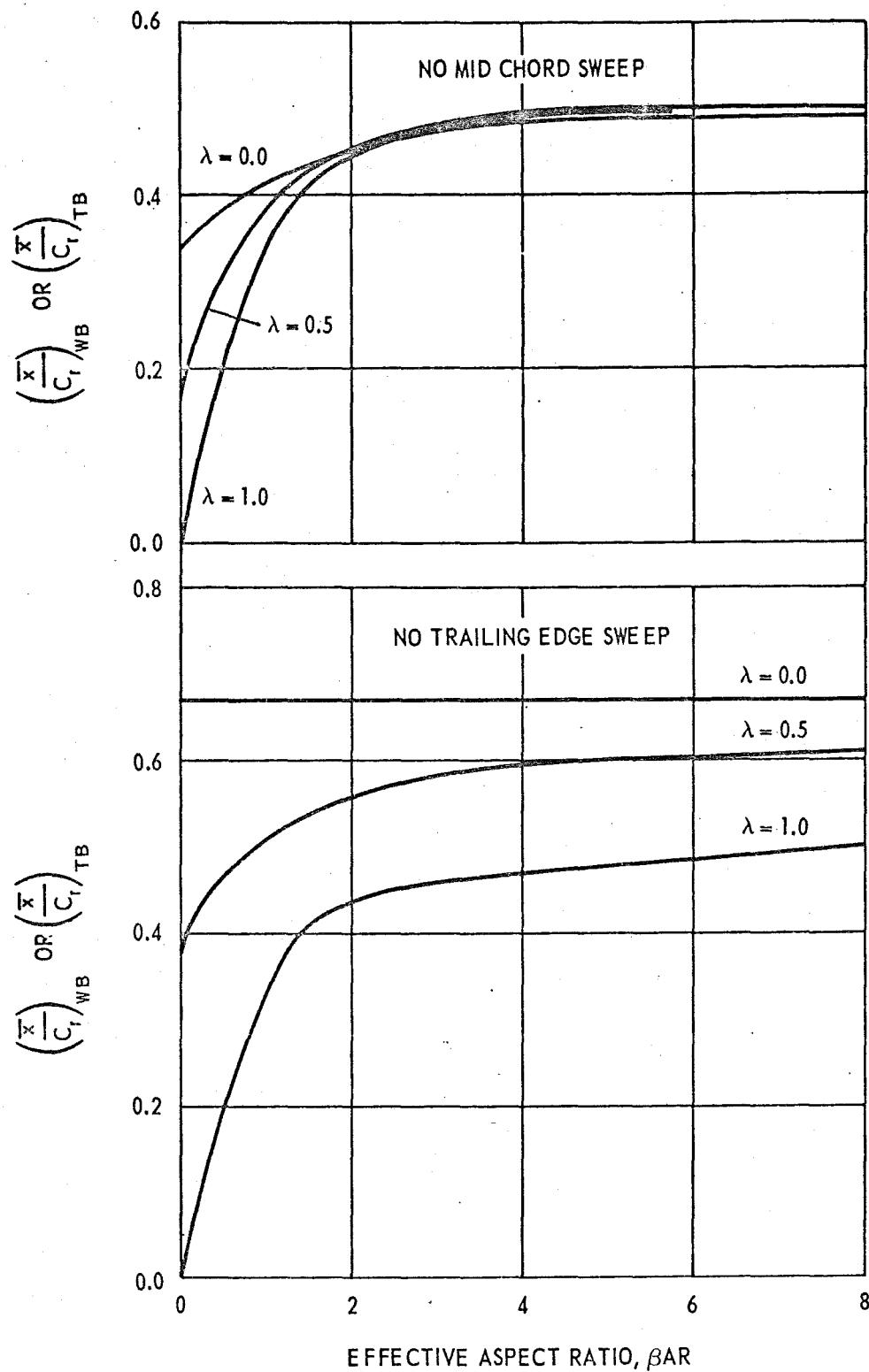


Figure 17b — $M > 1.0$

Figure 18 – Subsonic Center of Pressure Location of the Lift on the Body in the Presence of Wings or Tails (from Reference 1)

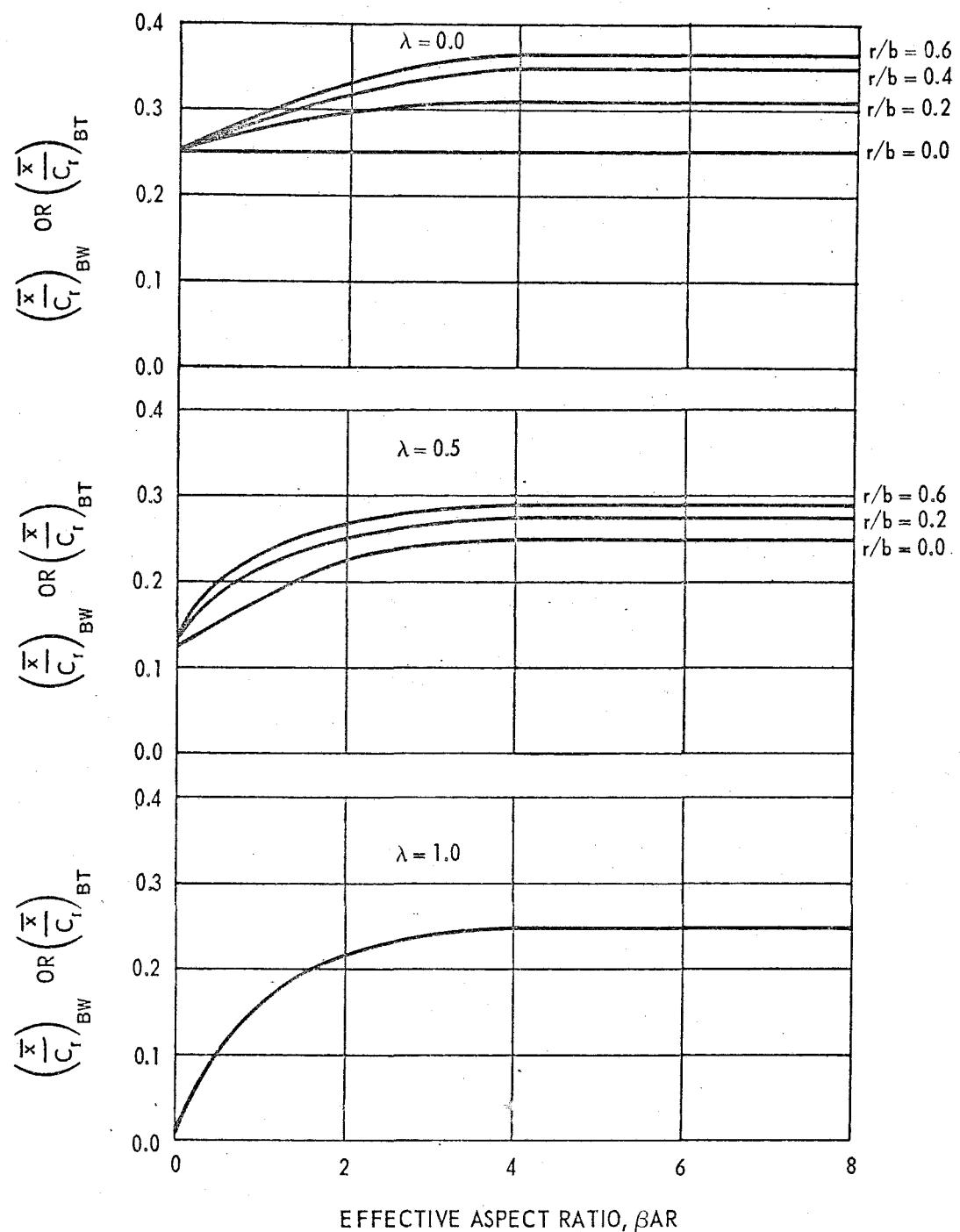


Figure 18a — No Mid Chord Sweep

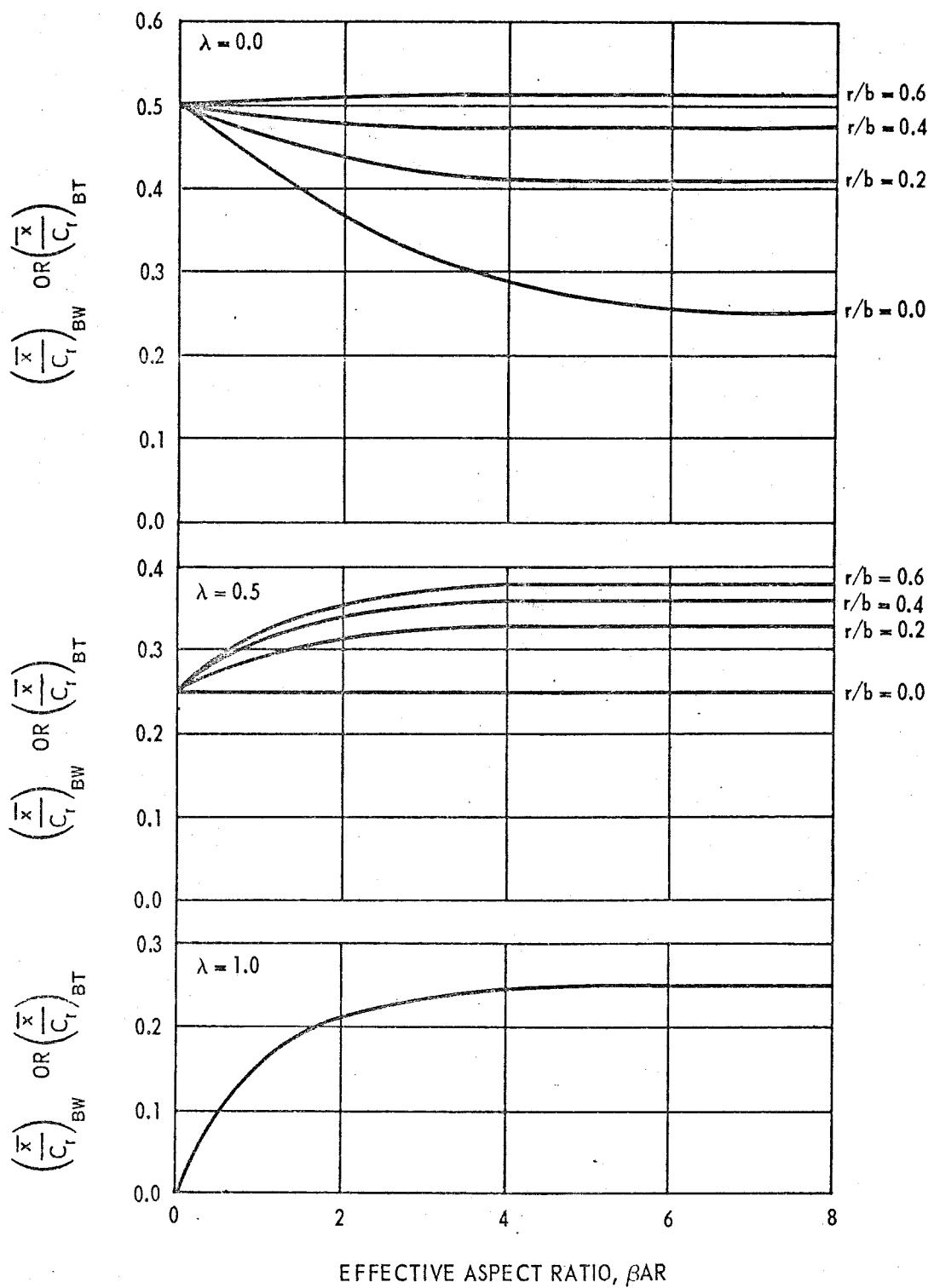


Figure 18b — No Trailing Edge Sweep

Figure 19 – Supersonic Center of Pressure Location of Lift on the Body in the Presence of Wings or Tails for $\beta \text{AR} (1 + \lambda) (1 + 1/m\beta) \leq 4.0$ (from Reference 1)

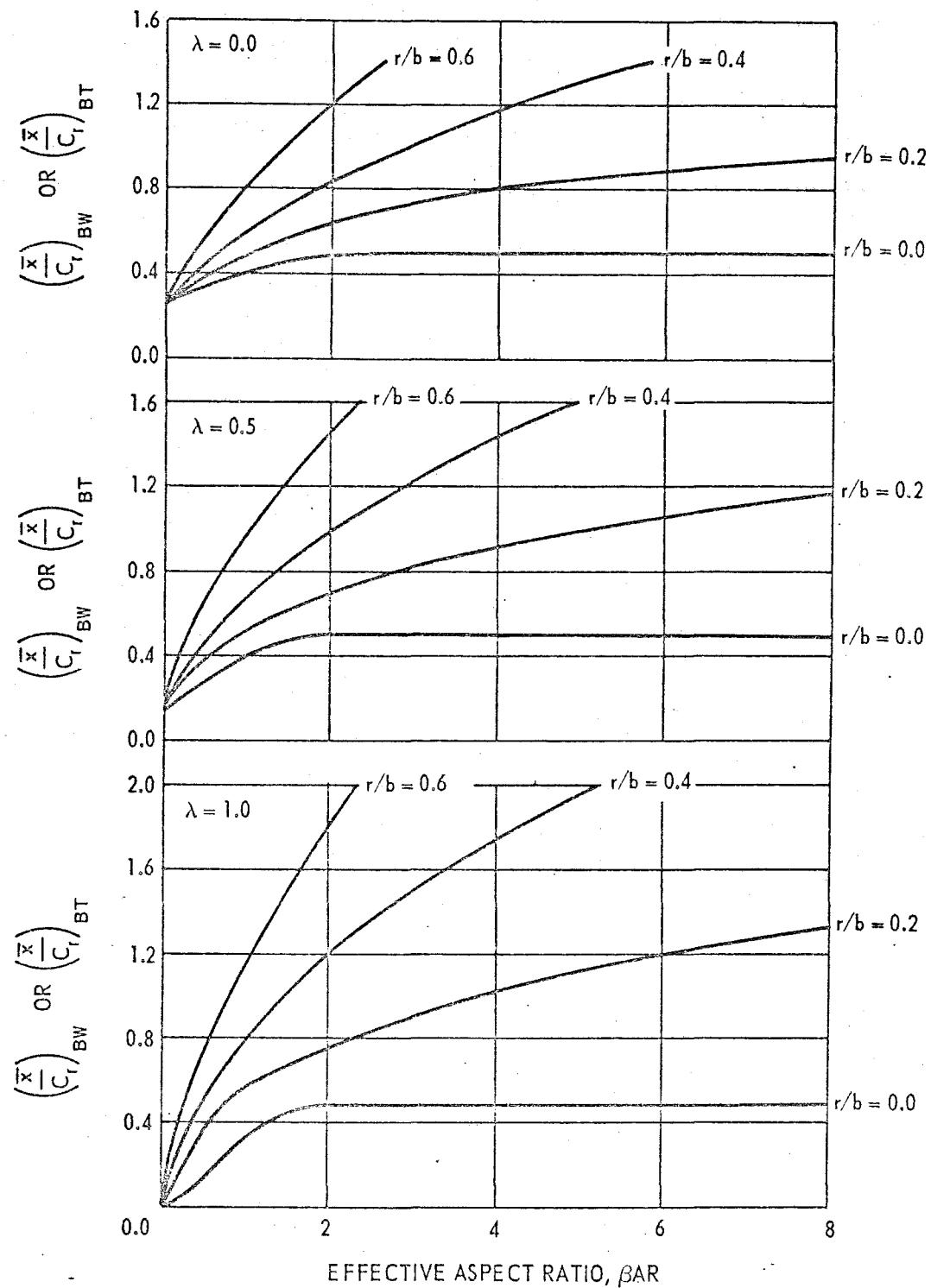


Figure 19a – No Mid Chord Sweep

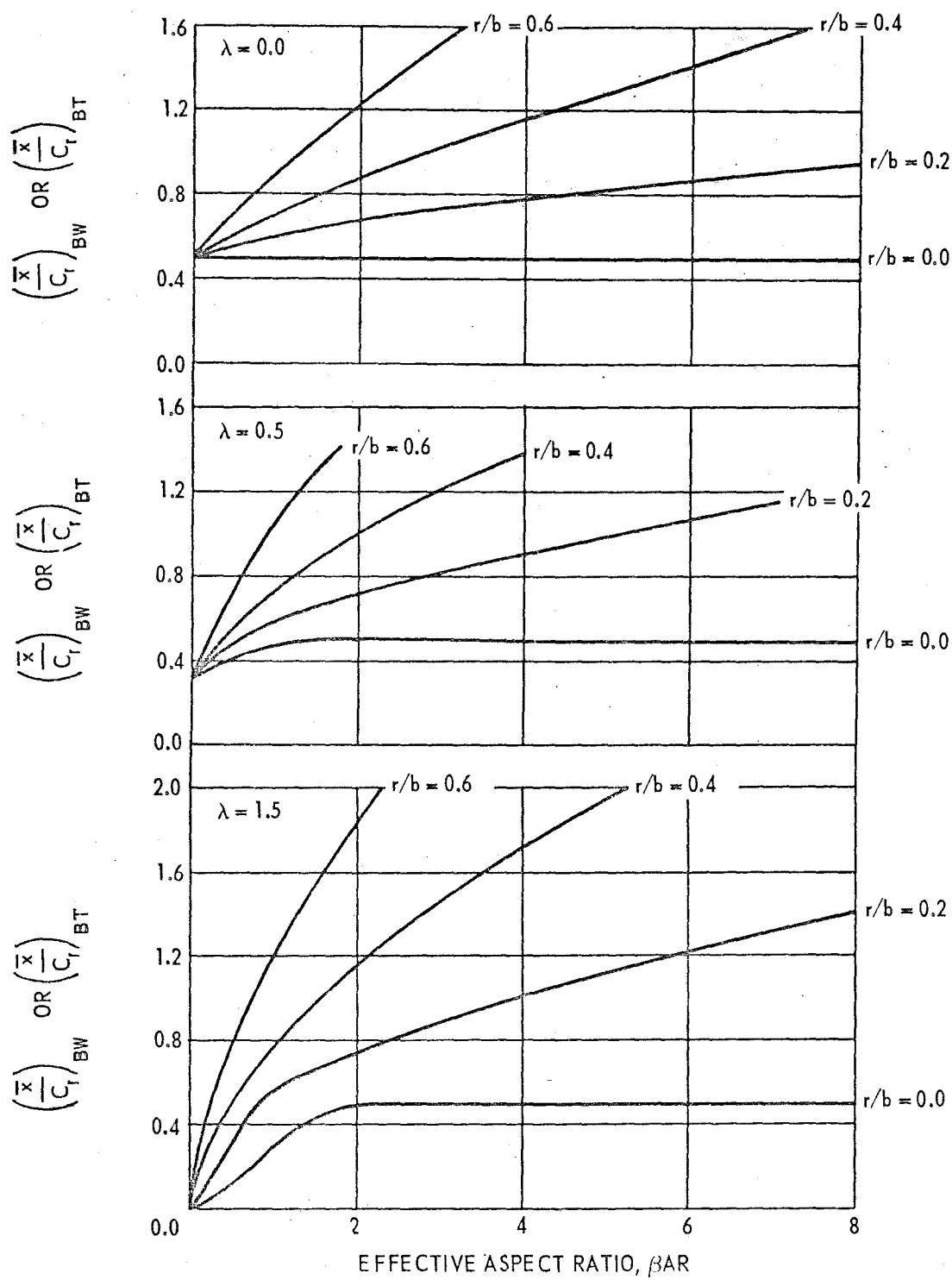


Figure 19b -- No Trailing Edge Sweep

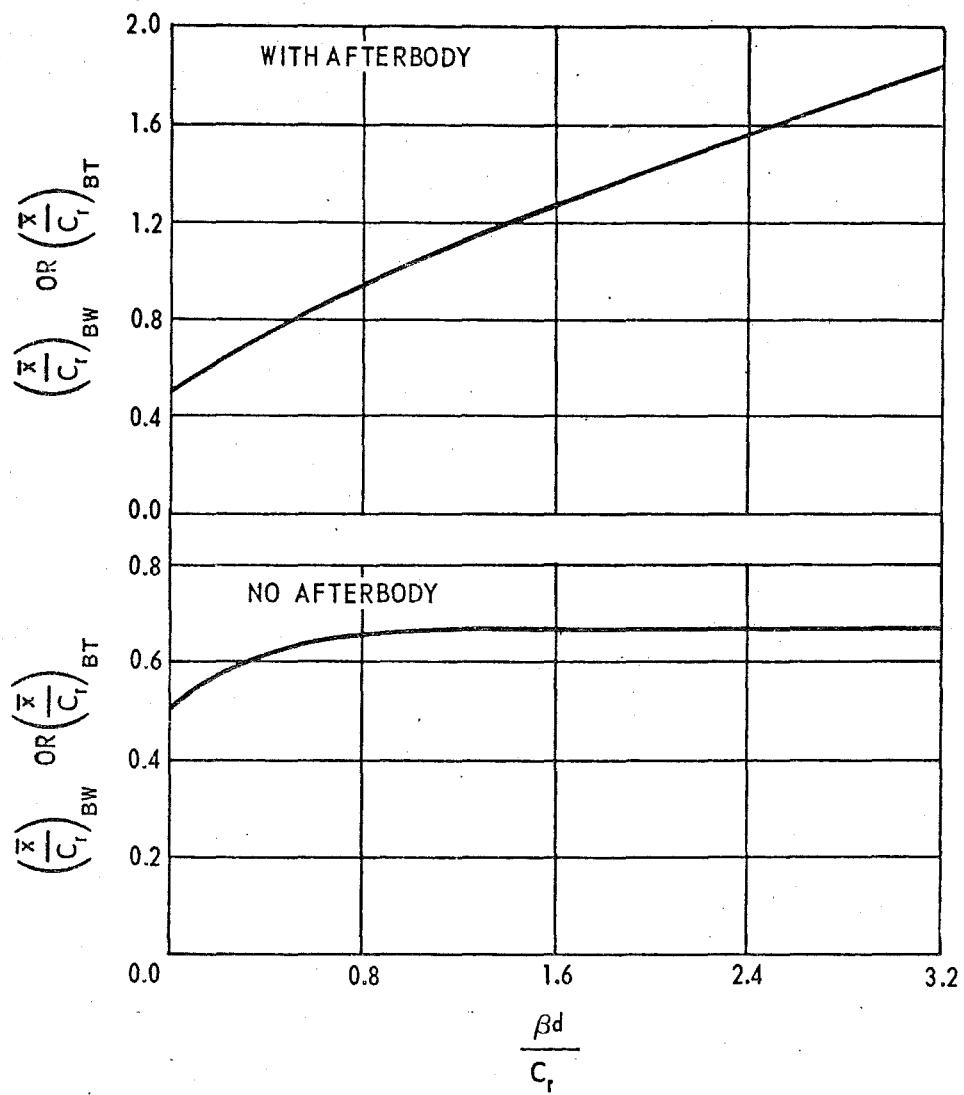


Figure 20 – Supersonic Center of Pressure Location of Lift on the Body in the Presence of Wings or Tails for $\beta \text{AR}(1 + \lambda)(1 + 1/m\beta) > 4.0$ (from Reference 1)

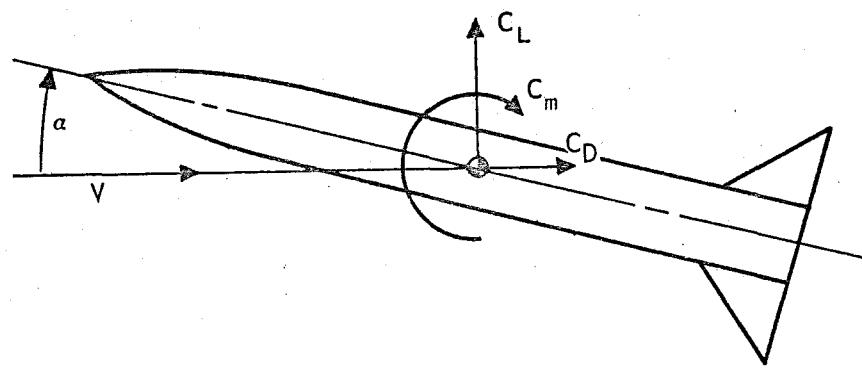


Figure 21a — Stability Axes

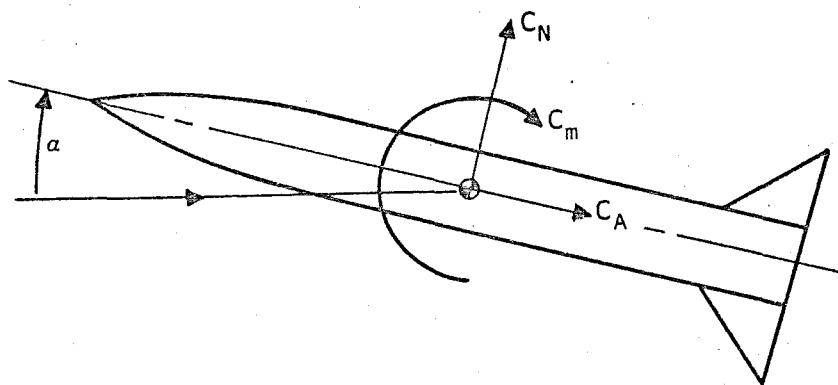
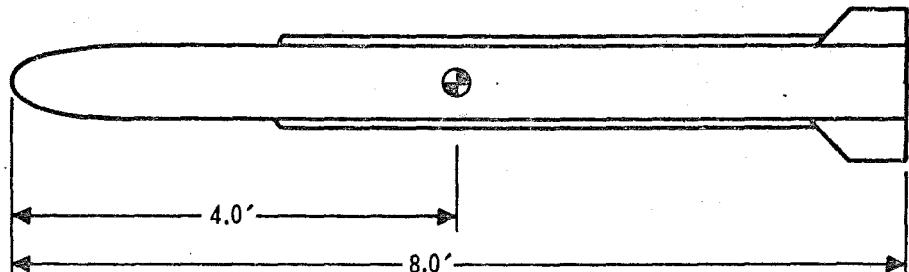
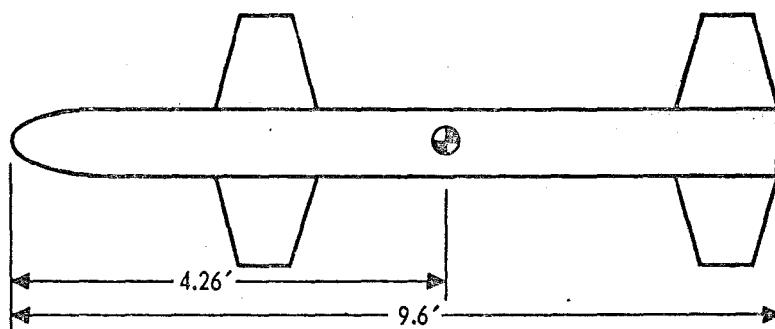


Figure 21b — Body Axes

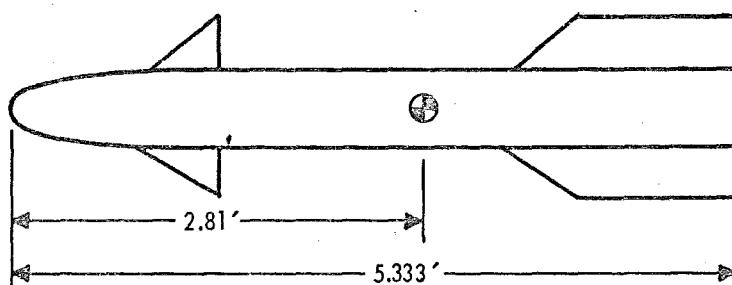
Figure 21 — Missile Axis Systems



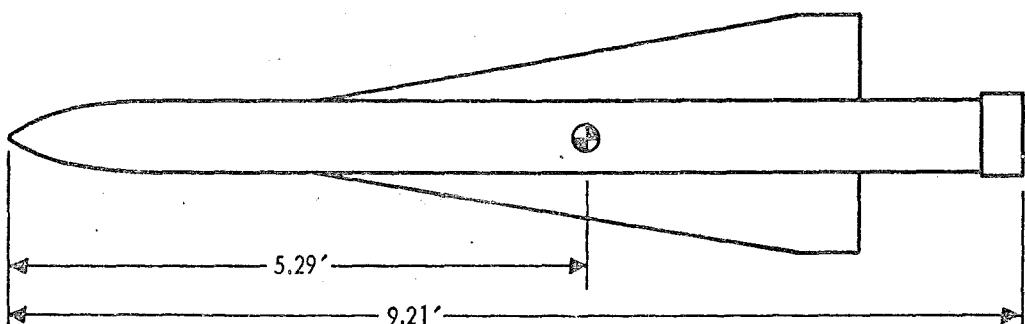
CONFIGURATION 1



CONFIGURATION 2



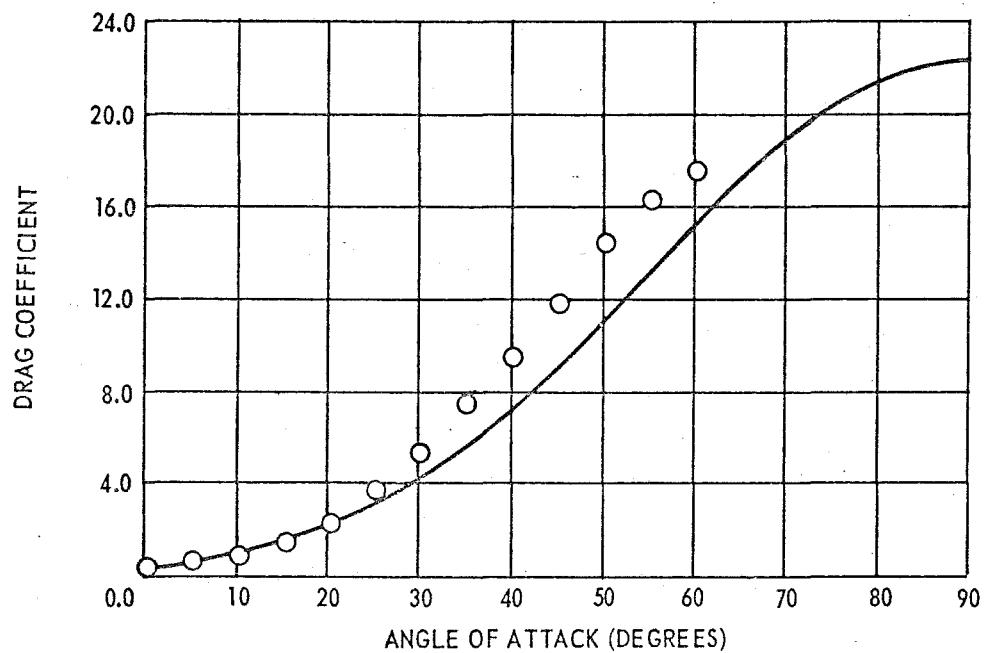
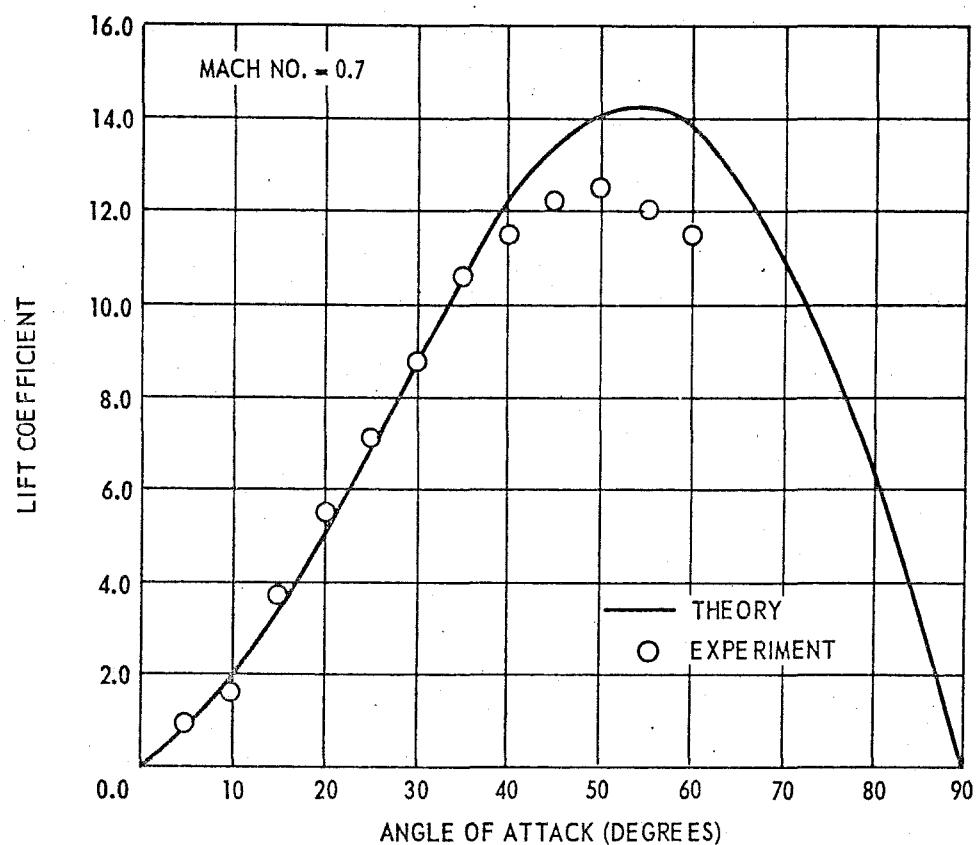
CONFIGURATION 3



CONFIGURATION 4

Figure 22 – Configurations Used to Compare Theory with Experiment

Figure 23 -- Comparison of Experimental Data with Theoretical Results for Configuration 1



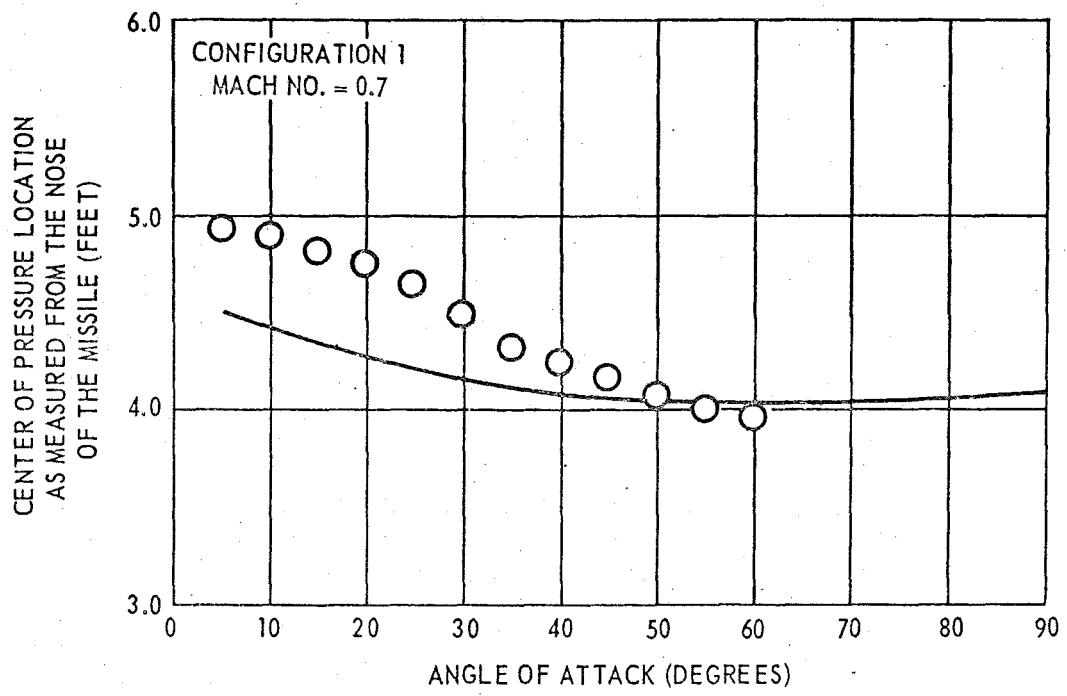


Figure 28 (Continued)

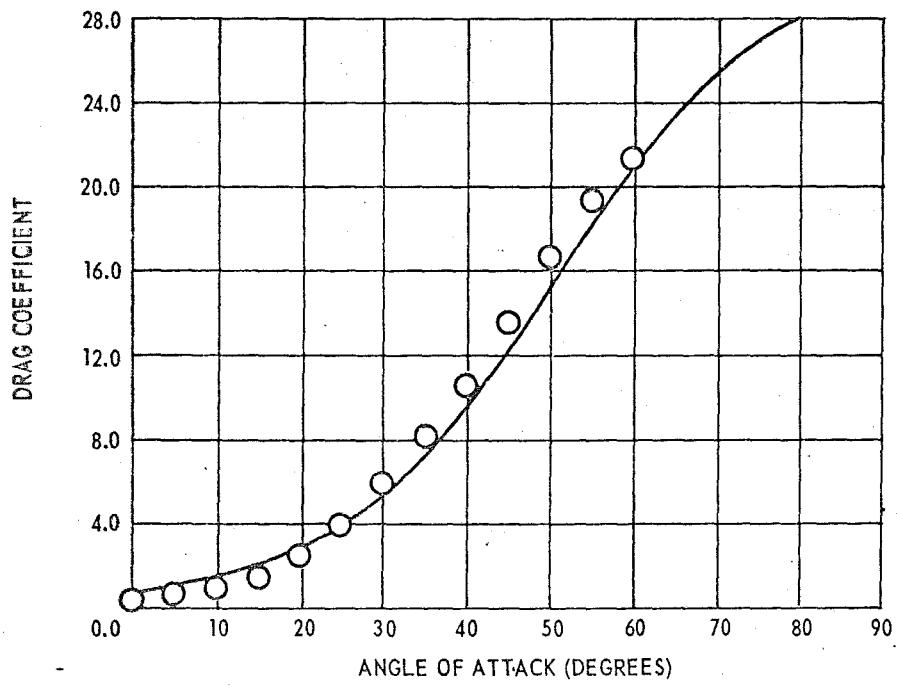
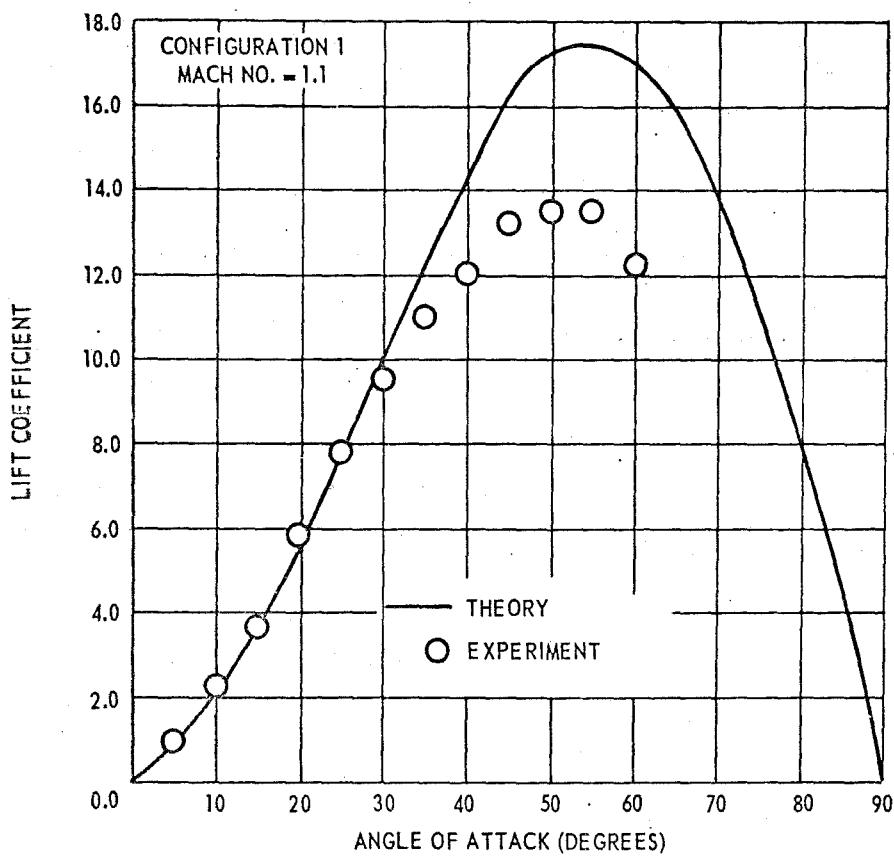


Figure 23 (Continued)

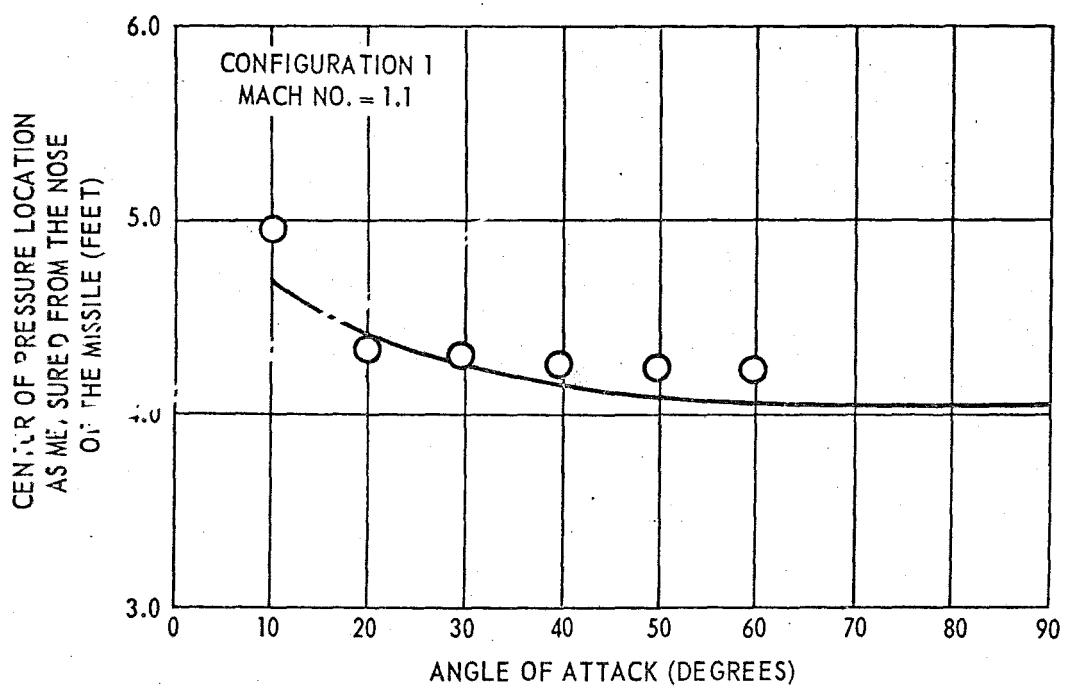
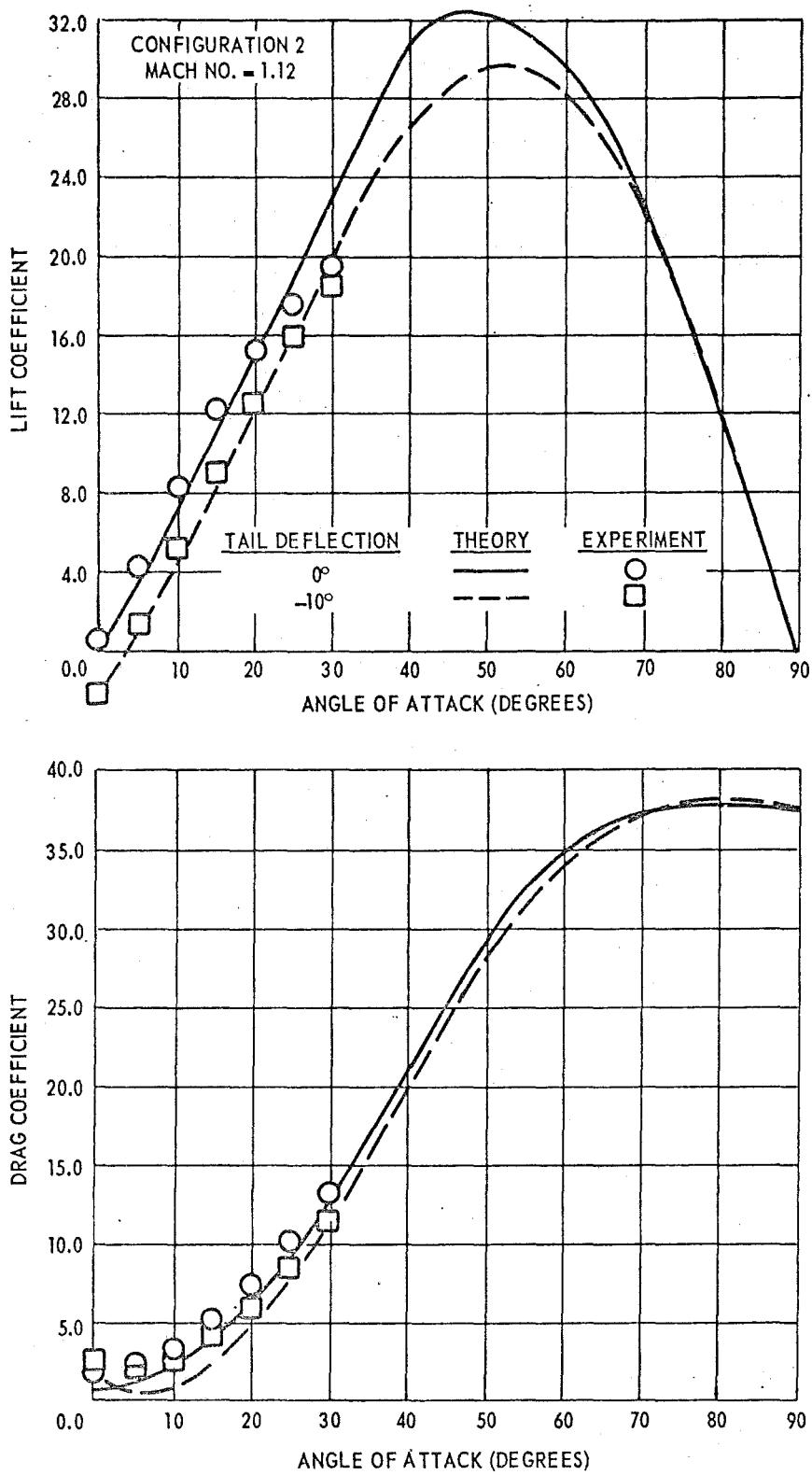


Figure 23 (Continued)

Figure 24 – Comparison of Experimental Data with Theoretical Results for Configuration 2



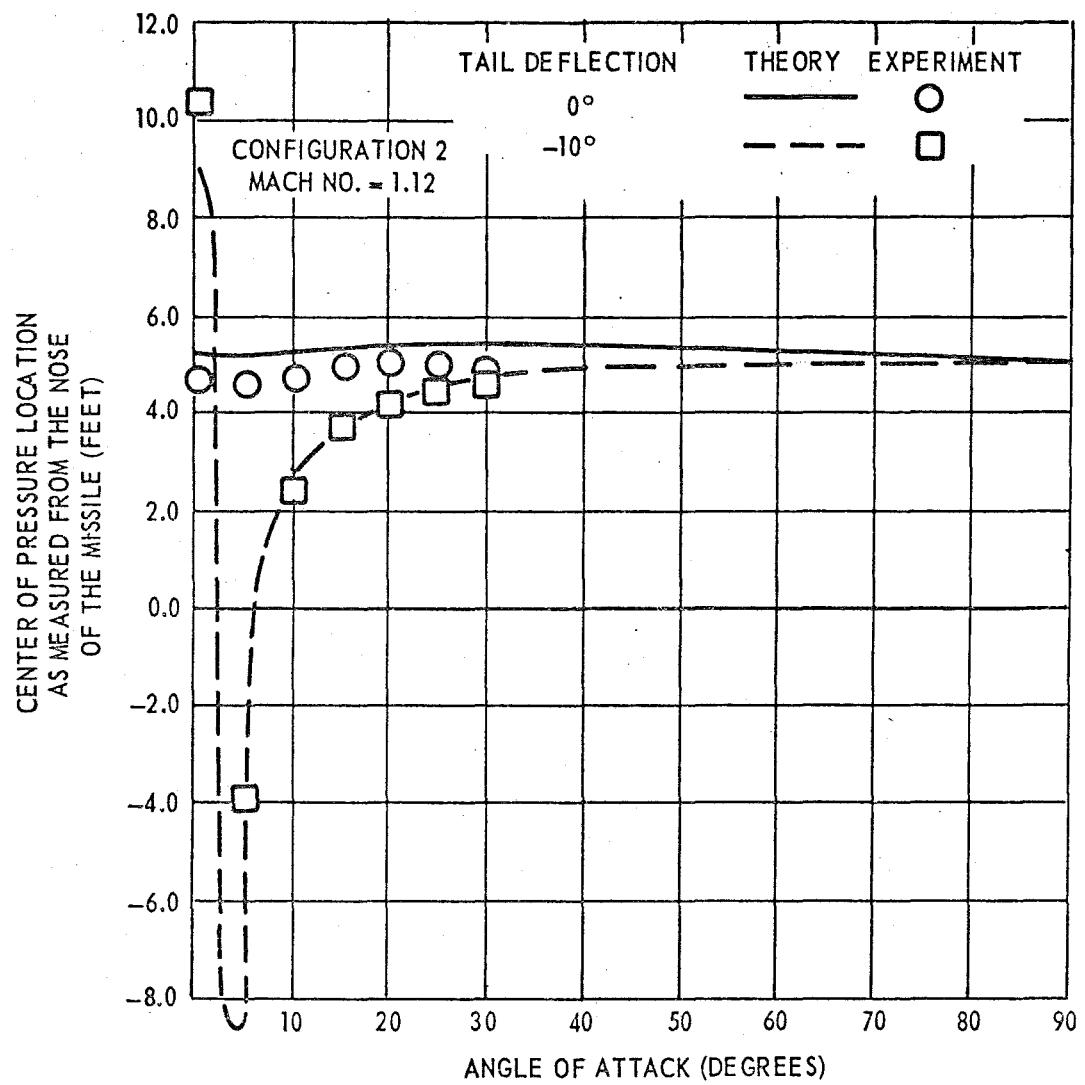


Figure 24 (Continued)

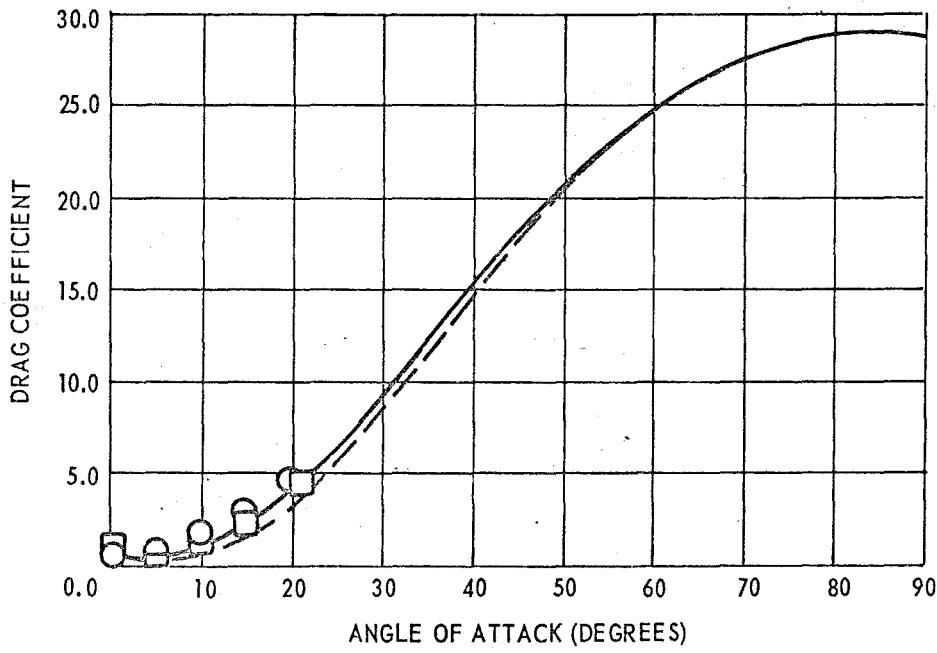
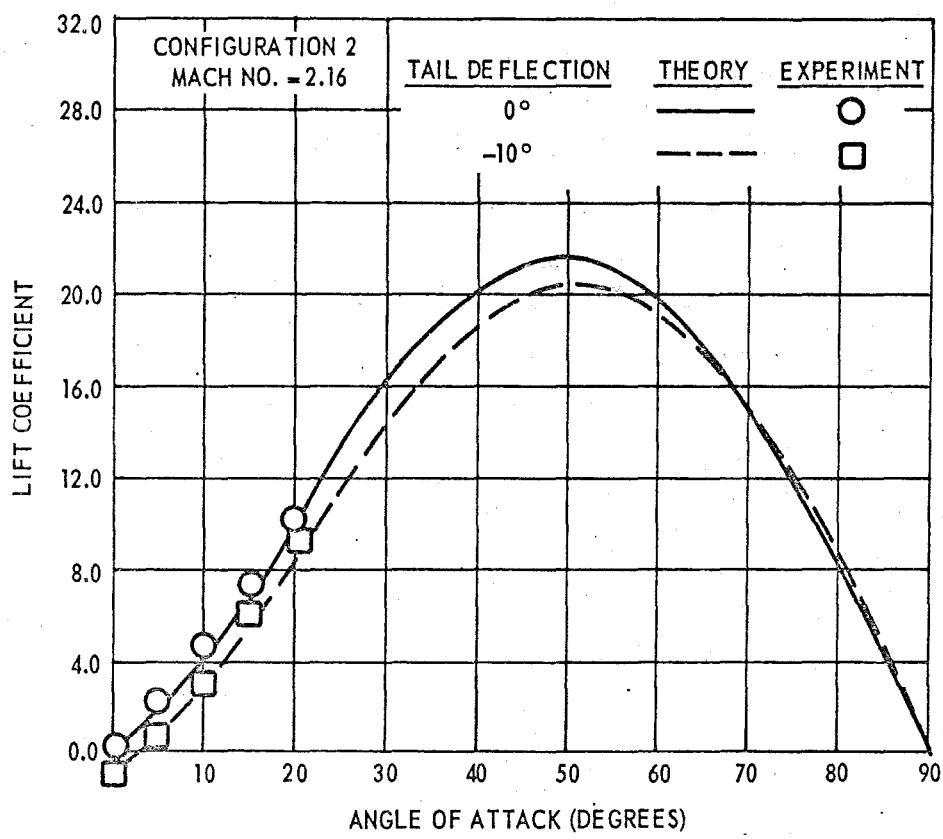


Figure 24 (Continued)

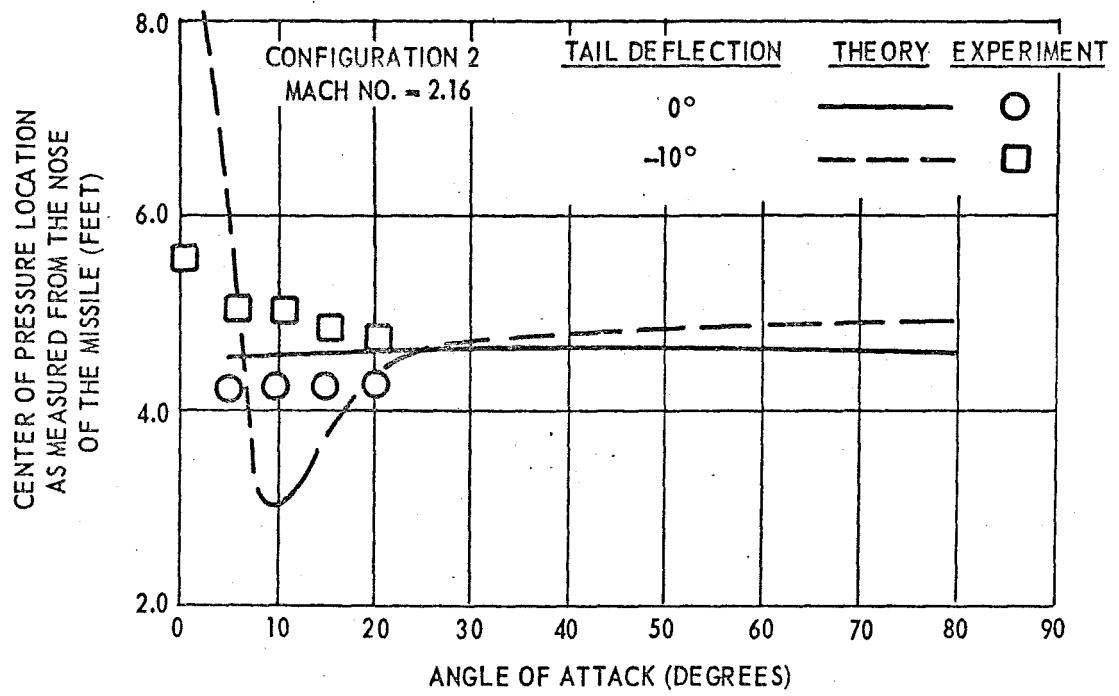
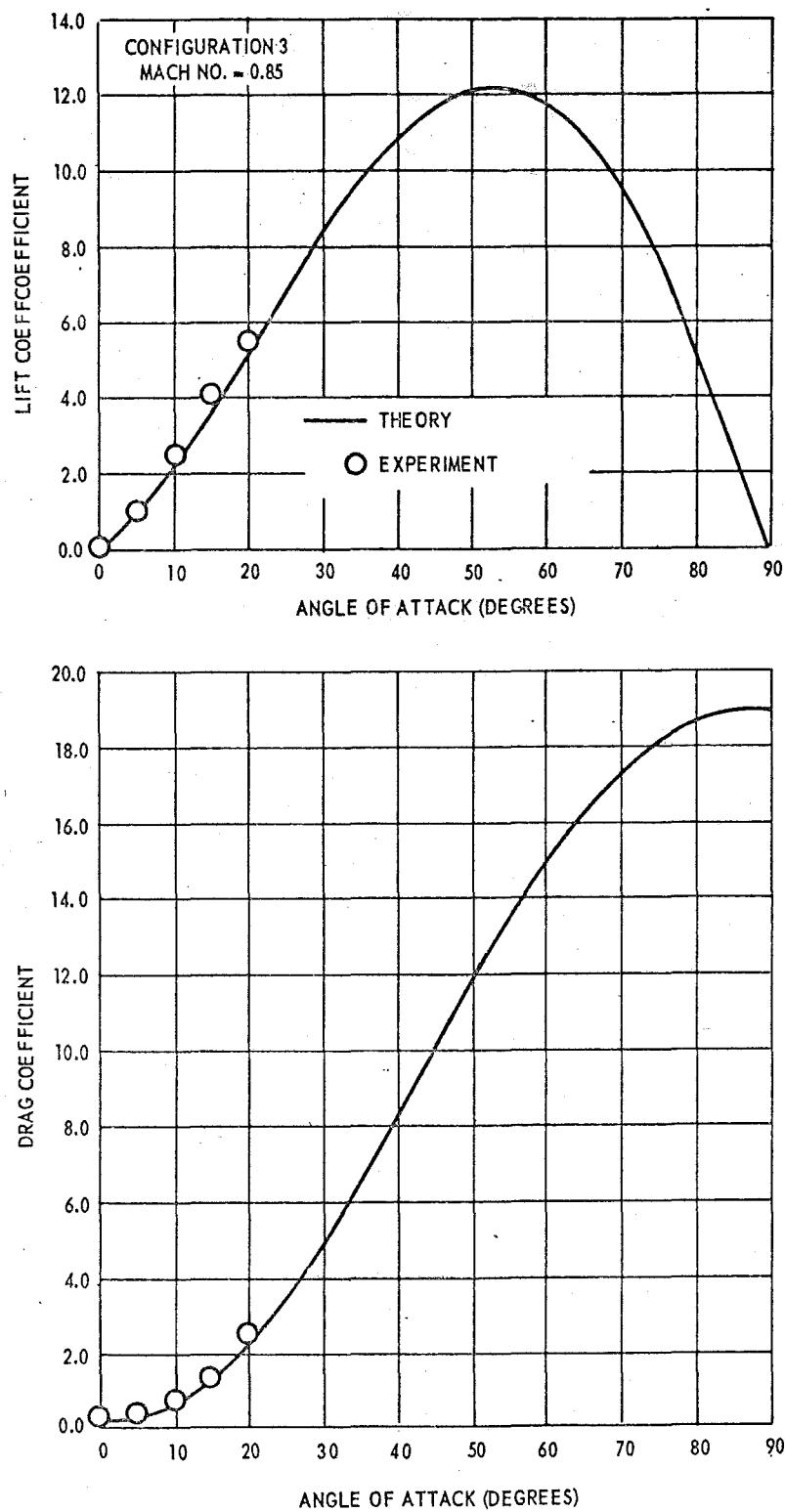


Figure 24 (Continued)

Figure 25 – Comparison of Experimental Data with Theoretical Results for Configuration 3



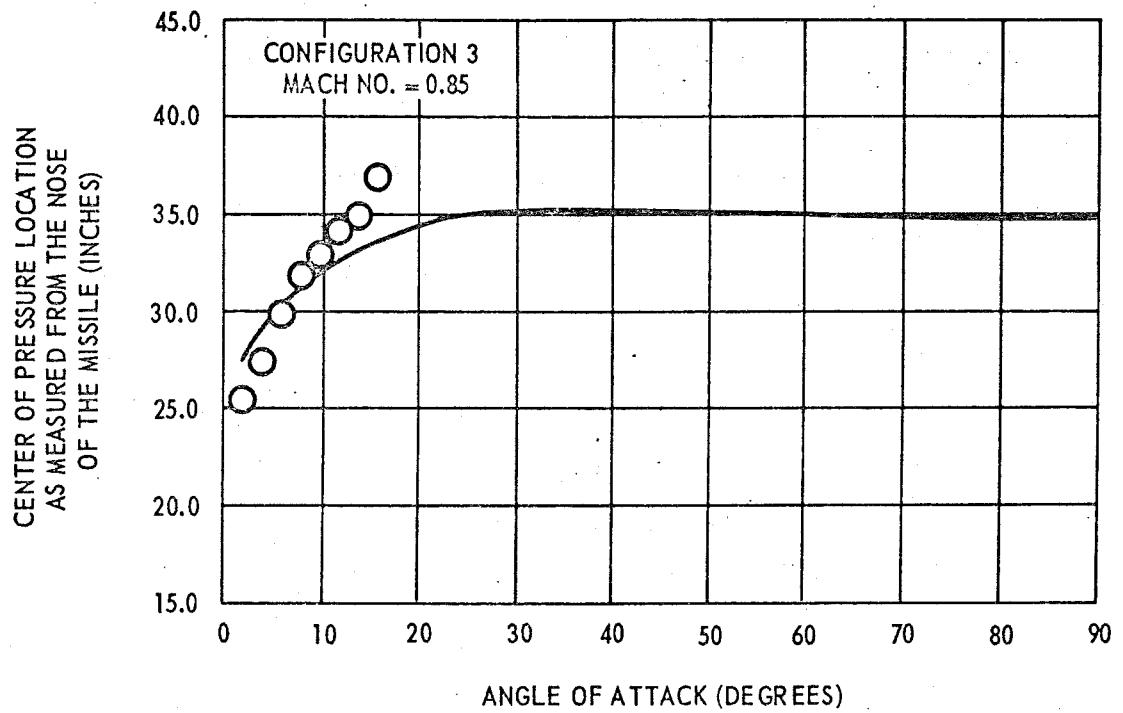


Figure 25 (Continued)

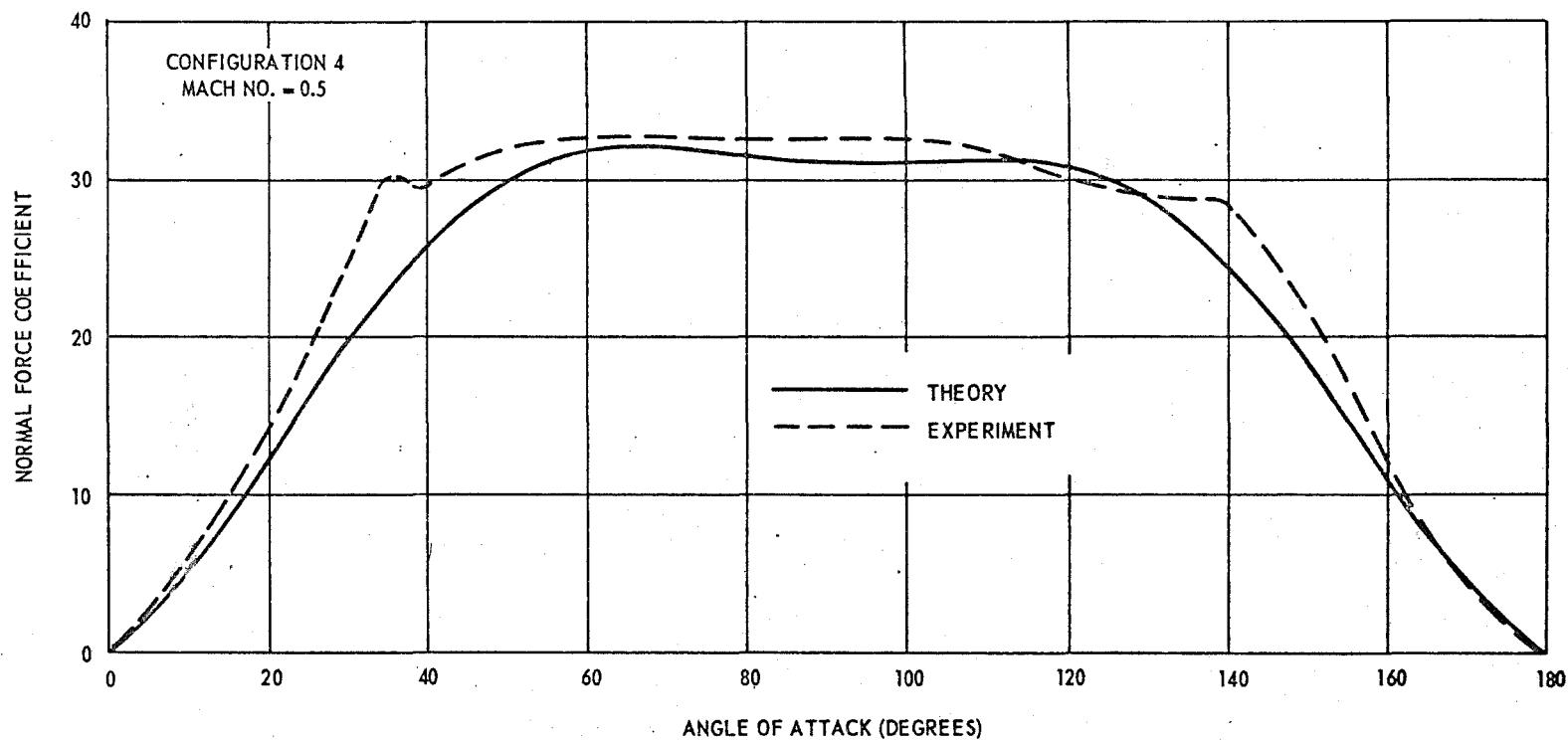


Figure 26 – Comparison of Experimental Data with Theoretical Results for Configuration 4

Table 1
COMPUTER PROGRAM LISTING

```

DIMENSION XVXM[16],XDT[16],XAL[48]
COMMON XVXM,XDT,XAL
COMMON CN,CA,CNB,CNW,CNT,CNW2,CLW,CLT,CLW2,CLI,CLWB,CLVISW,CLIT,
1      CLIW,CNTD,CAB,XCP2,XCG2,XCPB,XCPT,XCPW,XCPW2,XCG,XCPTV,
2      XLAMT4
COMMON LLKK,ISWPW,ISWPW2,IAFBW,IAFBT,IAFBW2,IL,LLLL,IJ,J,
1      II,NB8DY,IZZY,ICSC,INSE,NM,NMLK,IDL,IM,IAL,ISWP1,IAFB
COMMON XLAMW,XLAMT,XLAMW2,XMACW,XMACT,XMACW2,CLAMW,CLAMT,CLAMW2,
1      BW,BT,BW2,CR88TW,CR88TT,CR88W2,SW,ST,SW2,XWING,XTAIL,XWING2
2      ,XL,D,D1,XLN,AREA,XREF,SSUBS,XLB,B,ZF,ART,ARW,ARW2,XLN8SE
COMMON CLALM,BCLALM,CR88T,R1,CLAL1,XLAM1,BAR,RAT18,XKTB,XKBT,
1      XKWB,XKBW,XBCRBW,XKWBW,XKWBWT,XKBWW,XKBWT,XKBWW2,
2      XCPBW,XCPBT,XCPBW2,YCPBW,XCPBT,XCPBW2,XKB1,BDC,BDCW,BDCT,
3      BDCW2,CLALW,CLALT,CLALW2,REFT,BETA,AL,T8NST,H8NST,HT,XKTB1,
4      VXM,VXMR1,DELTA,XKCRWB,XBCRWB,XMAC,XLAMW4,XLAM24,XLAM2,
5      XLAM4,T8VC,T8VCCT,T8VCW2,T8VCW,EXS,ST8T,SWTB,SW2TB,RE,
6      CD8,CD8W,CD8T,CD8B,CD8W2,CLBW,CLWB2,CLBW2,CLTR,CLBT,CLTD,
7      CLTDB,CLBTD,CLVIST,CLVIW2,CMB,CD8WT
COMMON DC8SW,DC8ST,DC8S2,DC8W,DC8T,DC8W2
COMMON CDALZ
      IL = 0
3040 FORMAT[2I5,7F10.5]
3050 FORMAT[6I5]
3020 FORMAT[7F10.3]
3C10 FORMAT[10A6]
3110 FORMAT[1H1,10A6]
3120 FORMAT[6X,2HHT, 9X,2HD ,8X,2HXL,6X,6HXL8SE,5X,3HXCG,6X,4HAREA,6X,
14HXREF]
3C21 FORMAT[//,5X,5HT8VCW,5X,6HT8VCW2,5X,5HT8VCCT]
3140 FORMAT[ 3X, 15, 5X, 15, 4X, 7F15.6,/] ]
3150 FORMAT[4X,15,5X,15,5X,15,5X,15,6X,15,5X,15]
3240 FORMAT[ 6X,5HISWPW, 5X,6HIAFBW, 10X, 5HXLAMW, 10X, 5HCLAMW,
1 10X,5H BW ,9X,6HCR88TW,10X,5H SW ,10X,5HXMACW,10X,5HXWING,/] ]
3244 FORMAT[ 6X,6HISWPW2, 4X,6HIAFBW2, 9X, 6HXLAMW2, 9X, 6HCLAMW2,
1 10X,5H BW2 ,9X,6HCR88W2,10X,5H SW2 ,9X,6HXMACW2,9X,6HXWING2,/] ]
3248 FORMAT[ 6X,5HISWP1, 5X,5HIAFBT, 10X, 5HXLAMT, 10X, 5HCLAMT,
1 10X,5H BT ,9X,6HCR88TT,10X,5H ST ,10X,5HXMACT,10X,5HXTAIL,/] ]
3241 FORMAT[//,6X,5HICSC ,5X,5HINSE,5X,5HNELT,5X,5HNMACH,5X,5HNALPH,
15X,5HN8DY]
3333 READ 3010, TITL1,TITL2,TITL3,TITL4,TITL5,TITL6,TITL7,TITL8,TITL9,
1 TITL0:
      READ 3050, ICSC,INSE,IDL,IM,IAL,NB8DY
      READ 3040, ISWPW,IAFBW,XLAMW,CLAMW,BW,CR88TW,SW,XMACW,XWING
      READ 3040, ISWPW2,IAFBW2,XLAMW2,CLAMW2,BW2,CR88W2,SW2,XMACW2,XWIN
1G2
      READ 3040, ISWP1,IAFBT,XLAMT,CLAMT,BT,CR88TT,ST,XMACT,XTAIL
      READ 3020, HT,D,XL,XLN8SE,XCG,AREA,XREF
      READ 3020, T8VCW,T8VCW2,T8VCCT

```

```

PRINT 3110, TITL1,TITL2,TITL3,TITL4,TITL5,TITL6,TITL7,TITL8,TITL9,
1TITL0
PRINT 3241
PRINT 3150, ICSC,IN8SE, IDT, IM, IAL,NB8DY
PRINT 3240
PRINT 3140, ISWPW,IAFBW,XLAMW,CLAMW,BW,CR88TW,SW,XMACW,XWING
PRINT 3244
PRINT 3140, ISWPW2,IAFBW2,XLAMW2,CLAMW2,BW2,CR88W2,SW2,XMACW2,
1XWING2
PRINT 3248
PRINT 3140, ISWPT,IAFBT,XLAMT,CLAMT,BT,CR88TT,ST,XMACT,XTAIL
PRINT 3120
PRINT 3020, HT,D,XL,XLN8SE,XCG,AREA,XREF
PRINT 3021
PRINT 3020, T8VCW,T8VCW2,T8VCT
IL = 1 + IL
LLKK=0
LLLL=0
XCG2 = XCG
IZZY = 0
CALL GE8SUB
READ 4000, [XDT[M],M=1, IDT]
READ 4000, [XVXM[N],N=1, IM]
READ 4000, [XAL[NA],NA=1, IAL]
RE=REFT*VXM
VXM=XVXM[1]
D8 6002 IJ=1,IM
DELT A1=XDT[1]
D8 6001 II=1, IDT
ALPHA=XAL[1]
4000 FORMAT[16F5.1]
5000 F9RMAT[1H1,4HVXM,F5.2,2X,6HDELT A1,F6.2,/] 
5001 F9RMAT[2X,2HAL,3X,5HCLTB,2X,5HCDTAT,2X,4HCLWT,3X,4HCLTT,3X,3HCLB,
14X,3HCLI,4X,3HCNW,4X,3HCNT,4X,4HCNTD,3X,3HCNB,5X,2HCN,5X,2HCA,4X,
24HXCPW,3X,4HXCP T,3X,4HXCPB,3Y,4HXCP2,4X,2HCM,/] 
8200 PRINT 5000, VXM,DELT A1
PRINT 5001
DELT A=DELT A1/57.29578+0.00000001
D8 6000 J=1,IAL
AL=ALPHA/57.29578+0.0000001
1 VXR1=VXM
IZZY=IZZY+1
IF [IZZY = 4] 6666,6666,1111
6666 VXM=.6
1111 CALL CLASUB
IF[LLLL=1] 900,942,980
900 IF[IZZY=4] 6009,6009,925
925 CALL CATSUB
6009 XLAM14=ATAN[.5*(B1-D)*1./C8LAM+.25*XLAM1*CR88T+.25*CR88T]/
1[.5*(B1-D)]]
RE=REFT*VXM*XMAC
IF [RE=1.E06] 6010,6020,6020

```

601C AA=•0835
 XNN=•.211
 Gθ Tθ 6070
 602C IF [RE=1•E07] 6030,6040,6040
 603C AA=•052
 XNN=•.177
 Gθ Tθ 6070
 604C IF [RE=1•E08] 6050,6060,6060
 605C AA=•0333
 XNN=•.1488
 Gθ Tθ 6070
 606C AA=•0221
 XNN=•.127
 607C CF=AA*RE**XNN
 CDθ=4•*CF*[1•+2•*TθVC+100•*TθVC*•4•]
 EXS*[D•C]/[8•*CθLAM]
 IF [ISWP1=1] 6080,6080,6090
 608C EXS=2•*EXS
 609C EXS=[CRE8T*D/2•*EXS]*2•
 IF [IZZY=4] 6091,6091,6092
 6091 IF [IZZY=3] 6093,6094,6095
 6092 IF [AL] 2401,2402,2402
 2401 PDC=•6DC
 2402 LLKK=LLKK+1
 IF [LLKK=2] 2403,2404,2420
 2403 IF [SW] 2410,2410,2420
 241C LLKK=LLKK+1
 2404 IF [SW2] 2411,2411,2420
 2411 LLKK=LLKK+1
 2420 IF [LLKK=2] 930,943,950
 930 XKWBW=XKWB
 XKWBW=XKWB
 XCPWB=XWING+XBCRWB*CR8AT
 XCPWB=XWING+XBCRRW*CR8AT
 BDCW=8DC
 96C CLW=SIN[AL]*[XKWBW+XKBW]*CLALW*SW*COS[AL]/AREA
 CLWB=SIN[AL]*XKWBW*CLALW*SW*COS[AL]/AREA
 CLBW=CLW=CLWB
 CLVISW=[SIN[AL]*SIN[AL]*SW*COS[AL]/AREA]*8DCW
 CLW=CLW=CLVISW
 6093 CD8W=CDE*[SW+EXS]/AREA
 XLAMW4=XLAM14
 TθVCW=TθVC
 SWTθT=SW+EXS
 IZZY=IZZY+1
 IF [SW2] 942,942,511
 511 CLAM=COS[CLAMW2]/SIN[CLAMW2]
 BCCLAM=BETA*CCLAM
 CR8BT=CRE8W2
 B1=BW2
 IAFB=IAFBW2
 CLAL1=CLALW2
 XLAM1=XLAMW2
 TθVC=TθVCW2
 XMAC=XMACW2
 ISWP1=ISWPW2

BAR=BETA*ARW2
 RATI0=CR00T/[BETA*D]
 IF [IZZY=4] 6009,6009,925
 943 XKWBW2=XKWB
 XKBWW2=XKBW
 XCPWB2=XWING2+XBCRBW*CR00T
 XCPBW2=XWING2+XBCRBW*CR00T
 PDCW2=PDC
 944 CLW2=SIN[AL]*[XKWBW2+XKBWW2]*CLALW2*SW2*COS[AL]/AREA
 CLWB2=SIN[AL]*XKWBW2*CLALW2*SW2*COS[AL]/AREA
 CLBW2=CLW2-CLWB2
 CLVIW2=[SIN[AL]*SIN[AL]*SW2*COS[AL]/AREA]*8DCW2
 CLW2=CLW2+CLVIW2
 6094 CDEW2=CDB*[SW2+EXS]/AREA
 XLAM24=XLAM14
 XLAM2=XLAM14
 SW2T@T=SW2+EXS
 942 IZZY = IZZY + 1
 LLKK=LLKK+2
 IF [ST] 980,980,940
 94C CBLAM=COS[CLAMT]/SIN[CLAMT]
 ART=[BT*D]**2/ST
 BCBLAM=BETA*CBLAM
 CR00T=CR00TT
 B1=BT
 BAR=BETA*ART
 CLAL1=CLALT
 IAFB=IAFBT
 XMAC=XMACT
 TBVC=T9VCT
 ISWP1=ISWP1
 XLAM1=XLAMT
 RATI0=CR00T/[BETA*D]
 IF [IZZY=4] 6009,6009,925
 950 XKWBT=XKWB
 XKBWT=XKBW
 XCPBT=XTAIL+XBCRBW*CR00T
 PDC=8DC
 951 CLT=[XKWBW2+XKBWT]*SIN[AL]*CLALT*ST*COS[AL]/AREA
 CLTB=SIN[AL]*XKWBW2*CLALT*ST*COS[AL]/AREA
 CLBT=CLT-CLTB
 CLTDB=XKTB*CLALT*SIN[DELTA]*ST*COS[AL+DELTA]/AREA
 CLTD=[XKTB+XKBW2]*CLALT*SIN[DELTA]*ST*COS[AL+DELTA]/AREA
 CLBDT=CLTD-CLTDB
 CLVIST=[SIN[AL+DELTA]*SIN[AL+DELTA]]*ST*COS[AL+DELTA]/AREA]*8DC
 CLT=CLT+CLVIST
 6095 CDB=CD8*[ST+EXS]/AREA
 STTB=ST+EXS
 XLAMT4=XLAM14
 IF [IZZY=4] 1610,1610,6098
 6098 XCPTB=XTAIL+[[XKWBW2*SIN[AL]*XBCRBW+XKTB*SIN[DELTA]*XBCRBW]/
 1[XKWBW2*SIN[AL]+XKTB*SIN[DELTA]]]*CR00TT
 980 IF [IZZY = 4] 1610, 1610, 1710
 1710 XL0B = XL/D
 ZXm=VXM*ABS[SIN[AL]]
 IF [ZXm=0.8] 1310,1350,1350
 1310 CDC=2.4-SQRT[1.5129-1.5129*ZXM*ZXM]
 98 TB 1391

135C IF [ZXM-1.15] 1380,1370,1370
 138C CDC=1.6+SQRT[.0344-[ZXM-.975]**2]
 G9 T0 1391
 137C IF [ZXM-3.] 1360,1381,1381
 136C CDC=1.9-SQRT[.361-.09*[ZXM-3.]**2]
 G8 T0 1391
 1381 CDC=1.3
 1391 ETA=[0.0000475*[XL0B**3]]-[0.00173*[XL0B**2]]+[0.0298*XL0B]+0.5146
 IF [VXM-.5] 1395,1395,1392
 1392 IF [VXM-1.4] 1393,1394,1394
 1393 ETA=ETA+[1.-ETA]*[VXM-.5]*1.111111111111
 G9 T0 1395
 1394 ETA=1.
 1395 IF [XL0B-10.] 1320,1330,1340
 132C XK2K1=0.0054*[XL0B**2]+0.104*XL0B+C+437
 G8 T0 1600
 133C XK2K1=0.939
 G8 T0 1600
 134C XK2K1=0.939+[0.001525*[XL0B-10.0]]
 160C ALP=AL
 IF [AL] 1602,1601,1601
 1602 CDC=-CDC
 1601 CNB=[XK2K1*SIN[2.*ALP]*COS[ALP/2.]]*3.14159*D*D/[4.*AREA]
 1+ETA*CDC*[[XL*D]/AREA]*[[SIN[ALP]]**2]
 XC=XCG/D
 CM81=[XK2K1*X0*SIN[2.*ALP]*COS[ALP/2.]]*3.14159*D*D*D/
 1[4.*AREA*XREF]
 CM82=[ETA*CDC*[[XL*D]/AREA1*[[XCG-[XL/2.]]/D]*[[SIN[ALP]]**2]]]
 1*D/XREF
 CM8=CM81+CM82
 161C RE=REFT*VXM*XL
 IF [RE-1*E06] 1611,1612,1612
 1611 AA=+0835
 XNA=-.211
 G9 T0 1617
 1612 IF [RE-1*E07] 1613,1614,1614
 1613 AA=+052
 XNA=-.177
 G8 T0 1617
 1614 IF [RE-1*E08] 1615,1616,1616
 1615 AA=+033
 XNA=-.1488
 G9 T0 1617
 1616 AA=+0221
 XNA=-.127
 1617 CFB0=AA*RE**XN
 CDBB=1.02*CFB0*D*[1.+1.5/[XL0B**1.5]+7.0/[XL0B**3.]]*SSUBS/AREA
 IF [IZZY-4] 1618,1618,1619
 1618 VXM=1.199999
 IF [SW] 1620,1620,1621
 1620 CDBW=0.
 1621 IF [SW2] 1622,1622,1623
 1622 CDBW2=0.
 1623 IF [ST] 1624,1624,1625
 1624 CDBT=0.
 1625 FBNST=CDBW+CDBW2+CDBT+CDBB

GO TO 1639
 1619 IF [VXM=1.2] 1631,1631,1632
 1631 IF [VXM=.8] 1638,1638,1639
 1638 CDALZ=CD8B
 CD8WBT=CD8B+CD8W+CD8T+CD8W2
 GO TO 1663
 1639 IF [SW]=1700,1700,1701
 1700 DCD8SW = 0.0
 SWTET=0.0
 CD8W = 0.0
 GO TO 1702
 1701 $XXM = VXM * \text{SQRT}[\cos(XLAMW4)]$
 $\text{SQMITC} = \text{SQRT}[\text{ABS}[(XXM * XXM) - 1.0]] / [T8VCW ** 0.33333]$
 $\text{ATC} = \text{ARW} * [T8VCW ** 0.33333]$
 IZT=1
 1640 IF [ATC=1.5] 1641,1642,1642
 1642 IF [VXM=1.0] 1643,1643,1644
 1643 FUNCT=3.3081-1.88779*SQMITC+11.0916*SQMITC*SQMITC=18.6087*
 1SGMITC**3+7.4633*SQMITC**4
 GO TO 1650
 1644 FUNCT=3.4
 GO TO 1650
 1641 IF [ATC=.5] 1645,1645,1646
 1646 IF [VXM=1.0] 1647,1647,1648
 1647 FUNCT=2.47917-1.42798*SQMITC+.324405*SQMITC*SQMITC
 GO TO 1650
 1648 FUNCT=2.5*ATC
 GO TO 1650
 1649 IF [VXM=1.0] 1649,1649,1651
 1649 FUNCT=.5325-.47202*SQMITC+.08631*SQMITC*SQMITC
 GO TO 1650
 1651 FUNCT=.55917+.33333*SQMITC-.071429*SQMITC*SQMITC
 1652 IF [IZT=2] 1652,1653,1654
 1652 DCD8SW=FUNCT*[T8VCW ** 1.66667]*[(COS(XLAMW4)) ** 2.5]
 DCD8W=DCD8SW
 1702 IF [SW] 1703,1703,1704
 1703 DCD8S2 = 0.0
 SW2TET = 0.0
 CD8W2= 0.0
 GO TO 1705
 1704 IZT = 2
 $XXM = VXM * \text{SQRT}[\cos(XLAM24)]$
 $\text{SQMITC} = \text{SQRT}[\text{ABS}[(XXM * XXM) - 1.0]] / [T8VCW2 ** 0.33333]$
 $\text{ATC} = \text{ARW2} * [T8VCW2 ** 0.33333]$
 GO TO 1640
 1653 DCD8S2=FUNCT*[T8VCW2 ** 1.66667]*[(COS(XLAM24)) ** 2.5]
 DCD8W2=DCD8S2
 1705 IF [ST] 1706,1706,1707
 1706 DCD8ST = 0.0
 STTET=0.0
 CD8T = 0.0
 GO TO 1708
 1707 IZT = 3
 $XXM = VXM * \text{SQRT}[\cos(XLAMT4)]$
 $\text{SQMITC} = \text{SQRT}[\text{ABS}[(XXM * XXM) - 1.0]] / [T8VCT ** 0.33333]$
 $\text{ATC} = \text{ART} * [T8VCT ** 0.33333]$
 GO TO 1640

1654 $CDDBST = FUNCT * [T0VCT * 1.66667] * [COS[XLAMT4]] * 2.5$
 $CDDBT = CDBST$
 1708 $C8VC = 1.31213 * 0.36633 * VXM + .06038 * VXM * 2 + .00601 * VXM * 3 + .000275 * VXM * 4$
 $CDFPTR = 1.02 * CFB8D * C8VC * SSUBS / AREA$
 $CDPPTR = CD8B * 1.02 * CFB8D * SSUBS / AREA$
 $IF [VXM = 1.0] 1709, 1709, 1711$
 1711 $CDPPTR = [CDPPTR / 0.2] * [1.2 * VXM]$
 1709 $FR = XLN0SE / D$
 $CDWN1 = 0.000407 * [FR * 8] + 0.0102 * [FR * 7] + 0.108 * [FR * 6] + 0.616 * [FR * 5]$
 $+ 2.074 * [FR * 4] + 4.183 * [FR * 3] + 4.891 * [FR * 2] + 3.017 * FR + 0.7795$
 $CDWN2 = 0.000172 * [FR * 8] + 0.00453 * [FR * 7] + 0.050 * [FR * 6] + 0.304 * [FR * 5]$
 $+ 1.096 * [FR * 4] + 2.406 * [FR * 3] + 3.160 * [FR * 2] + 2.391 * [FR] + 1.000$
 $CDWN3 = .000125 * [FR * 8] + .00370 * [FR * 7] + .0447 * [FR * 6] + .288 * [FR * 5]$
 $+ 1.076 * [FR * 4] + 2.385 * [FR * 3] + 3.141 * [FR * 2] + 2.529 * FR + 1.300$
 $IF [VXM = 0.8] 1664, 1661, 1662$
 1661 $CDPTR = CDWN1$
 $GO TO 1658$
 1662 $IF [VXM = 1.0] 1655, 1665, 1666$
 1665 $CDPTR = CDWN2$
 $GO TO 1658$
 1655 $CDPTR = [CDWN2 - CDWN1] / 0.2 * [VXM = 0.8] + CDWN1$
 $GO TO 1658$
 1666 $IF [VXM = 1.2] 1667, 1668, 1664$
 1668 $CDPTR = CDWN3$
 $GO TO 1658$
 1667 $CDPTR = [CDWN3 - CDWN2] / 0.2 * [VXM = 1.0] + CDWN2$
 $GO TO 1658$
 1664 $CDPTR = 0.0$
 1658 $CD8B = CDFPTR + CDPPTR + CDPTR * VXM * 3 * 14159 * D * D / [4 * AREA]$
 $CDALZ = CD8B$
 $CD8T = 1.1 * [DCD8ST * [STT8T / AREA] + CD8T]$
 $CD8W = 1.1 * [DCD8SW * [SWT8T / AREA] + CD8W]$
 $CD8W2 = 1.1 * [CD8W2 + DCD8S2 * [SWT8T / AREA]]$
 $CD8WBT = 1.1 * [CD8W + DCD8SW * [SWT8T / AREA] + CD8W2 + DCD8S2 * [SWT8T / AREA]]$
 $+ CD8T + DCD8ST * [STT8T / AREA] + CD8B]$
 $IF [IZZY = 4] 1659, 1659, 1663$
 1659 $T8NST = CD8WBT$
 $H8NST = H8NST + ZF * [T8NST - H8NST]$
 $VXM = VXMRI$
 $IZZY = IZZY + 1$
 $GO TO 1$
 1632 $CD8WBT = [(H8NST - T8NST) / [SQRT[3.1] - SQRT[1.2]]] * SQRT[VXM] + T8NST$
 $+ [H8NST - T8NST] / [1 - SQRT[3.1] / SQRT[1.2]]$
 $CD8T = 1.1 * [DCD8T * [STT8T / AREA] + CD8T]$
 $CD8W = 1.1 * [DCD8W * [SWT8T / AREA] + CD8W]$
 $CD8W2 = 1.1 * [CD8W2 + DCD8W2 * [SWT8T / AREA]]$
 $CDAB = CD8WBT - CD8T - CD8W - CD8W2$
 $CDALZ = CD8B$
 1663 $CD8 = XK2K1 * SIN[2 * AL] * SIN[AL / 2] * 3 * 14159 * D * D / [4 * AREA] + ETA * CDC * XL$
 $+ D * [SIN[AL]] * 3 / AREA$
 $CLB = XK2K1 * [SIN[2 * AL]] * COS[AL / 2] * 3 * 14159 * D * D / [4 * AREA] + ETA * CDC * XL$
 $+ D * [SIN[AL]] * 2 * COS[AL] / AREA = CDALZ * COS[AL] * COS[AL] * SIN[AL]$
 $CAB = CD8WBT$
 $IF [AL] 1603, 1604, 1603$

1604 XCPB=0.0
 G9 T9 1605
 1603 XCPB=[[XCG/XREF]-[CMB/CNB]]*XREF
 1605 IF [SW] 1607,1607,1606
 1606 IF [ST] 2973,2973,1608
 1607 IF [ST] 1990,1990,2974
 1608 R=D/2.
 IF [ICSC = 1] 1970,1970,1971
 1971 XB1=BW/2.
 XB2=BT/2.
 TT=D/BT
 HW1=0.5*CR00TT*ABS[SIN[DE[TA]]+{XCPWB=XTAIL=CR00TT}*ABS[SIN[AL]]]
 XLAM1=XLAMW
 G9 T9 1969
 1970 XB1=BT/2.
 XB2=BW/2.
 TT=D/BW
 HW1=[XCPTB=XWING=CR00TN]*ABS[SIN[AL]]
 XLAM1=XLAMT
 1965 FTRT=[[XB2=R]/[2.*[1.-TT]]]*[[3.14159/4.]-[[3.14159*TT**2]/4.]-TT+
 1[[[1.+TT**2]**2]/[2.*[1.+TT**2]]]*ARSIN[[1.+TT**2]/[1.+TT**2]]]
 FW=FTRT+R
 FI1=[FW*R**2]/[[FW**2]+[HW1**2]]
 H1=[HW1*R**2]/[[FW**2]+[HW1**2]]
 ZC=FW
 ZD=HW1
 ZLT=0.0
 D0 1800 I=1,4
 ZL1=[[XB1-[XLAM1*R]]-[ZC*[1.-XLAM1]]]/[2.*[XB1=R]]
 ZL2=ALBG[[ZD**2]+[[ZC-XB1]**2]]/[ZD**2]+[[ZC-R]**2]]
 ZL3=[[1.-XLAM1]/[XB1=R]]*[[XB1=R]+[ZD*ATAN[[ZC-XB1]/ZD]-ATAN[[ZC-
 1R]/ZD]]]]
 ZL=[ZL1*ZL2]-ZL3
 IF[I=2] 1810,1820,1820
 1810 ZC=ZC
 G9 T9 1850
 1820 IF[I=3] 1830,1840,1850
 1830 ZL=ZL
 ZC=FI1
 ZD=H1
 G9 T9 1850
 1840 ZL=ZL
 ZC=FI1
 1850 ZLT=ZLT+ZL
 1800 CONTINUE
 IF[ICSC=1] 2970,2970,2971
 2970 ART=[BT-D]**2/ST
 XBT=BT/2.
 CLI=[CLALW*CLALT*XKWB1*SIN[AL]]*SW*2,*ZLT*[XBT=R]]/[2.*3.14159*ART*
 1FTRT*AREA*[1.+XLAMT]]
 CLI=CLI*CeS[AL]
 CLT=CLT+CLI
 CLIT=CLI
 CLIW=0.
 XCPTV=XCPTB
 G9 T9 2972

2971 ARW=[BW-D]*2/SW
 XBW=BW/2.
 CLI=[CLALW*CLALT*[XKWB1*SIN[AL]+XKTB1*SIN[DELT]]*ST*2.+ZLT*[XBW-
 1R]]/[2.*3.14159*ARW*FTRT*AREA*[1.+XLAMW]]
 CLI=CLI*COS[AL]
 CLW=CLW+CLI
 CLIW=CLI
 CLIT=0.
 XCPTV=XCPWB
 GO TO 2972

2973 CLT=0.
 CLALT=0.
 CLTD=0.
 CDTD=0.
 CLBT=0.
 CLBDT=0.
 CLTB=0.
 CLTDB=0.
 CLVIST=0.
 CLIT=0.
 CLIW=0.
 CLI=0.
 GO TO 2972

2974 CLW=0.
 CLALW=0.
 CLBW=0.
 CLWB=0.
 CLVISW=0.
 CLIW=0.
 CLIT=0.
 CLI=0.

2972 IF [SW2] 700,700,701
 700 CLW2=0.
 CLALW2=0.
 CLBW2=0.
 CLWB2=0.
 CLV1W2=0.
 701 ALPHA=AL
 IF [VXM=0.5] 1975,1976,1976

1975 XK=1.17
 GO TO 1973

1976 IF [VXM=1.0] 1977,1978,1978
 1977 XK=2.0-SQRT[0.764-[VXM=0.126]**2]
 GO TO 1973

1978 IF [VXM=2.0] 1979,1979,1980
 1979 XK=2.0-SQRT[1.298-[VXM=2.13]**2]
 GO TO 1973

1980 XK=0.87
 1973 CDT=XK*ABS[SIN[AL]]*ST/AREA
 CDW=XK*ABS[SIN[AL]]*SW/AREA
 CDW2=XK*ABS[SIN[AL]]*SW2/AREA
 CDD=XK*ABS[SIN[AL+DELT]]*ST/AREA
 CDTD=CDD=CDT
 CATD=CDTD*COS[AL]-CLTD*SIN[AL]
 CNTD=CLTD*COS[AL]+CDTD*SIN[AL]
 CNT=CLT*COS[ALPHA]-CDT*SIN[ALPHA]
 CAT=CDT*COS[ALPHA]-CLT*SIN[ALPHA]
 CAW2=CDW2*COS[AL]-CLW2*SIN[AL]

$CAW = CDW * C0S[\text{ALPHA}] - CLW * S1N[\text{ALPHA}]$
 $CNW = CLW * C0S[\text{ALPHA}] + CDW * S1N[\text{ALPHA}]$
 $CLW2 = CLW2 * C0S[\text{AL}] + CLW2 * S1N[\text{AL}]$
 $CAB = CDB * C0S[\text{AL}] - CLB * S1N[\text{AL}]$
 $CA5 = CAT + CAW + CLW2 + CATD + CAB$

68 TR 1991

1990 $CN = CNB$
 $XCP2 = XCPB$
 $CA = CA5$
 $CLTBT = CLB$
 $CDTBT = CDB + CA * C0S[\text{AL}] * C0S[\text{AL}]$
 $CLALT = 0.$
 $CLALW = 0.$
 $CDT = 0.$
 $CDD = 0.0$
 $CDW = 0.$
 $CATD = 0.$
 $CNTD = 0.$
 $CNT = 0.$
 $CAT = 0.$
 $CAW = 0.$
 $CNW = 0.$
 $CLW = 0.$
 $CLT = 0.$
 $CLI = 0.$
 $CLTT = 0.$
 $CLWT = 0.$
 $CLALW2 = 0.$
 $CPW2 = 0.$
 $CAW2 = 0.$
 $CNTT = 0.$
 $CNWT = 0.$

$CNWT2 = 0.$

$CNW2 = 0.$
 $CLW2 = 0.$
 $CLWT2 = 0.$
 $XCPW2 = 0.$
 $XCPT = 0.$
 $XCPW = 0.$

68 TR 1992

1991 $CN = CNB + CNW + CNT + CNTD + CNW2$
 $CLTT = CLBT + CLTB + CLBDT + CLTDB + CLIT + CLVIST$
 $CLWT = CLBW + CLWB + CLIW + CLVISH + CLWB2 + CLWB2 + CLVISH2$
 $CNTT = [CLIT + CLTB + CLBDT + CLVIST] * C0S[\text{AL}] + CDT * S1N[\text{AL}] * CLTDB + CLBDT$
 $CNAT = [CLIW + CLWB + CLBW + CLVISH] * C0S[\text{AL}] + CDW * S1N[\text{AL}]$
 $CNWT2 = [CLWB2 + CLBW2 + CLVISH2] * C0S[\text{AL}] + CDW2 * S1N[\text{AL}]$
 $CLTBT = CLTT + CLWT + CLB$
 $CDTBT = CDB + CDW + CDD + CDW2 + CDWB$
 $CA = CDTBT * C0S[\text{AL}] - CLTBT * S1N[\text{AL}]$
 $IF [ST] 9901, 9901, 9902$

9902 $XCPT = [[CLTB * C0S[\text{AL}] + CDD * S1N[\text{AL}]] * XCPTB + CLBT * C0S[\text{AL}] * XCPBT + CLTDB * 1 * XCPTB + CLBDT * XCPBT + CLIT * C0S[\text{AL}] * XCPTB + CLVIST * C0S[\text{AL}] * XCPTB] / CNTT$
9901 $IF [SW] 9903, 9903, 9904$
9904 $XCPT = [[CLWB * C0S[\text{AL}] + CDW * S1N[\text{AL}]] * XCPWB + CLBW * C0S[\text{AL}] * XCPBW + CLIW * 1 * C0S[\text{AL}] * XCPWB + CLVISH * C0S[\text{AL}] * XCPWB] / CNWT$
 $IF [SW2] 9903, 9903, 1993$

1993 $XCPW2 = [CLWB2 * COS[AL] + CDW2 * SIN[AL]] * XCPWB2 + CLBW2 * COS[AL] * XCPBW2$
 1 + CLVIW2 * COS[AL] * XCPWB2] / CNWT2
 9903 IF [SW2] 1994, 1994, 1995
 1994 $XCPW2 = 0$.
 $CNWT2 = 0$.
 IF [SW] 1997, 1997, 1995
 1995 $XCPW = [XCPW] * CNWT + [XCPW2] * CNWT2] / [CNWT + CNWT2]$
 $CNWT = CNWT + CNWT2$
 IF [SW] 1997, 1997, 1998
 1997 $XCPW = 0$.
 1998 IF [ST] 1999, 1999, 1996
 1999 $XCPT = 0$.
 1996 $XCP2 = [CNB * XCPB + XCPT * CNTT + XCPW * CNWT] / [CNB + CNTT + CNWT]$
 1992 $CM = CN * [XCG2 * XCP2] / XREF$
 7001 $AL1 = AL * 57.29578$
 8700 PRINT 5002, AL1, CLT0T, CDT0T, CLWT, CLTT, CLB, CLI, CNWT, CNTT, CNTD, CNB,
 1CN, CA, XCPW, XCPT, XCPB, XCP2, CM
 5002 F8RFORMAT [1X, F4.0, 17[1X, F6.2]]
 ALPHA=XAL[J+1]
 6000 CONTINUE
 PRINT 9400, CLALT, CLALW, CLALW2
 9400 F8RFORMAT[/, 2X, 6HCLALT=, F6.3, 5X, 6HCLALW=, F6.3, 5X, 7HCLALW2=, F6.3]
 DELTA1=XDT[II+1]
 6001 CONTINUE
 VXM=XVXM[IJ+1]
 6002 CONTINUE
 IF [NR80DY=1L] 3334, 3334, 3333
 3334 CONTINUE
 STOP
 END

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SUBROUTINE GE0SUB
DIMENSION XVXM[16], XDT[16], XAL[48]
COMMON XVXM, XDT, XAL
COMMON CN, CA, CNB, CNW, CNT, CNW2, CLW, CLT, CLW2, CLI, CLWB, CLVISW, CLIT,
1     CLIW, CNTD, CAB, XCP2, XCG2, XCPB, XCPT, XCPW, XCPW2, XCG, XCPTV,
2     XLAMT4
COMMON LLKK, ISWPW, ISWPWT, ISWPW2, IAFBW, IAFBT, IAFBW2, IL, LLLL, IJ, J,
1     II, NBODY, IZZY, ICSC, INSE, NM, NMLK, IDT, IM, IAL, ISWP1, IAFB
COMMON XLAMW, XLAMT, XLAMW2, XMACW, XMACT, XMACW2, CLAMW, CLAMT, CLAMW2,
1     BW, BT, BW2, CR8BTW, CR8BT, CR8AW2, SW, ST, SW2, XWING, XTAIL, XWING2
2     , XL, D, D1, XLN, AREA, XREF, SSUBS, XL8B, ZF, ART, ARW, ARW2, XLN8SE
COMMON CLAL, BCCLAL, CR8BT, B1, CLAL1, XLAM1, BAR, RATIO, XKTB, XKBT,
1     XKWB, XKBW, XBCRBW, XKWBW, XKWBW2, XKWW, XKBT, XKBW2,
2     XCPBW, XCPBT, XCPBW2, XCPWB, XCPTB, XCPWB2, XKWBI, BDC, BDCW, BDC,
3     BDCW2, CLALW, CLALT, CLALW2, REFT, BETA, AL, T8NST, H8NST, HT, XKTB,
4     VXM, VXMR1, DELTA, XKCRWB, XBCRWB, XMAC, XLAMW4, XLAM24, XLAM2,
5     XLAM4, T8VC, T8VCT, T8VCW2, T8VCW, EXS, STTBT, SWTBT, SW2TBT, RE,
6     CDB, CDBW, CDAT, CD8B, CDAW2, CLBW, CLWB2, CLBW2, CLTB, CLBT, CLTD,
7     CLTDB, CLBDT, CLVIST, CLVIW2, CMB, CDBWT
COMMON DCD8SW, DCD8ST, DCD8S2, DCD8W, DCD8T, DCD8W2
COMMON CDALZ
DRAT = XLN8SE/D
D1=D
XL8B = XL/D
IF [CLAMW=5.] 10,10,20
10 CLAMW=5.
20 IF [CLAMW2=5.] 30,30,40
30 CLAMW2=5.
40 IF [CLAMT=5.] 50,50,60
50 CLAMT=5.
60 CONTINUE
CLAMW=CLAMW/57.29578
CLAMT=CLAMT/57.29578
CLAMW2=CLAMW2/57.29578
IF [INSE=1] 1571,1572,1571
1572 SS1=8.+3.*[DRAT=2.5]
GO TO 1579
1571 IF [INSE=2] 1573,1574,1573
1574 SS1=4.+8.*[DRAT=1.5]/3.
GO TO 1579
1573 IF [INSE=3] 1579,1576,1579
1576 SS1=4.+2.*[DRAT=1.87]
1579 SSUBS=SS1*[3.14159*D*D/4.]+3.14159*D*[XL-XLN8SE]
IF [DRAT=.6] 1580,1580,1581
1580 ZF=1.
GO TO 1594
1581 ZF=1.522*EXP[-.7*DRAT]
1594 Z=HT*0.001
IF [HT=35332.0] 1595,1595,1596
1595 T=519.=HT/280.
PS=[1.91-0.01315*Z]**5.256
GO TO 1597
1596 T=393.
B=1.69-0.0478*Z
PS=6.49*EXP[B]

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1597 C=49.1*SQRT[T]
PS=PS*70.9
RH0=PS/[1715.*T]
XMU=2.270*[T**1.5]/[[T+198.6]*[10.**8]]
REFT=[C*RH0]/XMU
RETURN
END

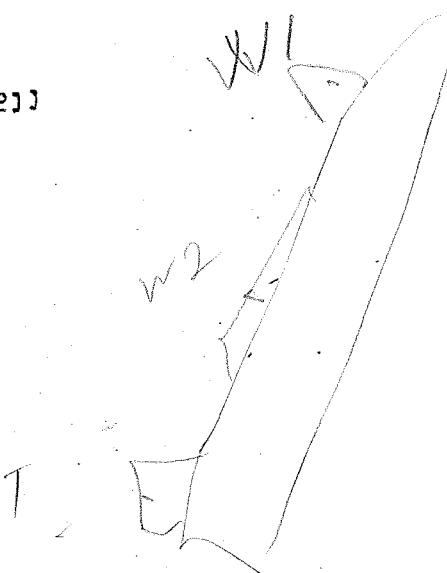
SUBROUTINE CLASUB
 DIMENSION XVXM[16], XDT[16], XAL[48]
 COMMON XVXM, XDT, XAL
 COMMON CN, CA, CNB, CNW, CNT, CNW2, CLW, CLT, CLW2, CLI, CLWB, CLVISW, CLIT,
 1 CLIW, CNTD, CAB, XCP2, XCG2, XCPB, XCPT, XCPW, XCPW2, XCG, XCPTV,
 2 XLAMT4
 COMMON LLKK, ISWPW, ISWPWT, ISWPW2, IAFBW, IAFBT, IAFBW2, IL, LLLL, IJ, J,
 1 II, NBODY, IZZY, ICSC, INSE, NM, NMLK, IDT, IM, IAL, ISWP1, IAFB
 COMMON XLAMW, XLAMT, XLAMW2, XMACW, XMACT, XMACW2, CLAMW, CLAMT, CLAMW2,
 1 BW, BT, BW2, CR00TW, CR00TT, CR00W2, SW, ST, SW2, XWING, XTAIL, XWING2
 2 , XL, D, D1, XLN, AREA, XREF, SSUBS, XL0B, ZF, ART, ARW, ARW2, XLN0SE
 COMMON CLAM, BCBLAM, CR00T, R1, CLAL1, XLAM1, BAR, RATIO, XKTB, XKBT,
 1 XKWB, XKBW, XBCRBW, XKWBW, XKWBT, XKWBW2, XKBWW, XKBWT, XKBWW2,
 2 XCPBW, XCPBT, XCPBW2, XCPWB, XCPBT, XCPWB2, XKWBI, BDC, BDCW, BDCD,
 3 BDCW2, CLALW, CLALT, CLALW2, REFT, BETA, AL, T0NST, H0NST, HT, XKTB1,
 4 VXM, VXM1, DELTA, XKCRWB, XBCRWB, XMAC, XLAMW4, XLAM24, XLAM2,
 5 XLAM4, T0VC, T0VCT, T0VCW2, T0VCW, EXS, STT0T, SWT0T, SW2T0T, RE,
 6 CDB, CDBW, CD0T, CD0B, CD0W2, CLBW, CLWB2, CLBW2, CLTB, CLBT, CLTD,
 7 CLTDB, CLBDT, CLVIST, CLVIW2, CMB, CD0WBT
 COMMON DCD0SW, DCD0ST, DCD0S2, DCD0W, DCD0T, DCD0W2
 COMMON CDALZ
 IF [VXM-1] 2, 2, 3
 2 BETA=SQRT [1 - VXM**2]
 G0 T0 4
 3 BETA=SQRT [VXM**2-1]
 4 IF [IZZY-4] 902, 902, 1111
 1111 IF [VXM-1] 41, 42, 41
 42 BETA=0.000001
 41 KFIN=0
 KFIN=KFIN+1
 IF [SW] 504, 504, 411
 411 ARW = [BW - D]**2/SW
 RAR=BETA*ARW
 ISWP=ISWPW ← set for why 1,
 XLAM=XLAMW
 AR=ARW
 505 IF [ISWP-1] 5, 5, 200
 5 IF [XLAM-25] 60, 10, 10
 1C IF [VXM-1] 20, 20, 30
 2C CLAR=-.1833*BAR+1.6
 G0 T0 370
 3C IF [BAR-1] 50, 40, 40
 4C CLAR=1.508*[1.26]**[2.-BAR]
 G0 T0 370
 5C CLAR=.3*BAR+1.6
 G0 T0 370
 6C IF [VXM-1] 70, 70, 80
 7C CLAR=-.1667*BAR+1.575
 G0 T0 370
 8C IF [BAR-1] 90, 90, 100
 9C CLAR=.1667*BAR+1.575
 G0 T0 370
 10C IF [BAR-2] 110, 110, 120
 11C CLAR=1.7417
 G0 T0 370

12C CLAR=1.428*[1.22]**[2.-BAR]
 G8 T8 370
 20C IF [XLAM=.1] 210,260,260
 21C IF [VXM=1.] 220,220,230
 22C CLAR=-.2077*BAR+1.575
 G8 T8 370
 23C IF [BAR=.25] 240,240,250
 24C CLAR=-.2077*BAR+1.575
 G8 T8 370
 25C IF [BAR=4.] 251,251,252
 251 CLAR=-.1668*BAR+1.667
 G8 T8 370
 252 CLAR=1.587*[1.26]**[2.-BAR]
 G8 T8 370
 26C IF [XLAM=.3] 270,320,320
 27C IF [VXM=1.] 280,280,290
 28C CLAR=-.2065*BAR+1.6
 G8 T8 370
 29C IF [BAR=0.75] 300,300,310
 30C CLAR=-.2065*BAR+1.6
 G8 T8 370
 31C IF [BAR=2.5] 311,311,312
 311 CLAR=-.217*BAR+2.293
 G8 T8 370
 312 CLAR=1.543*[1.26]**[2.-BAR]
 G8 T8 370
 32C IF [VXM=1.] 330,330,340
 33C CLAR=-.225*BAR+1.675
 G8 T8 370
 34C IF [BAR=1.] 350,350,360
 35C CLAR=-.225*BAR+1.675
 G8 T8 370
 36C CLAR=1.508*[1.26]**[2.-BAR]
 37C IF [AR=1.0] 800,800,810
 80C ARR=1.0
 G8 T8 820
 81C ARR=1./[AR**[[AR=1.]/AR]]
 82C CLAL=CLAR*ARR*AR
 IF [KFIN=2] 500,501,502
 50C CLALW=CLAL
 G8 T8 503
 501 CLALW2=CLAL
 G8 T8 503
 502 CLALT=CLAL
 503 IF [KFIN=2] 504,507,900
 504 KFIN=KFIN+1
 IF [SW2] 507,507,506
 506 AR=[BW2-D]**2/SW2
 BAR=BETA*AR
 ISWP=ISWPW2 Set for wing 2 o backlog start over
 ARW2=AR
 XLAM=XLAMW2
 G8 T8 505

```

5C7 KFIN=KFIN+1
    IF [ST] 509,509,508
5C8 AR=[BT-D]**2/ST
    BAR=BETA*AR
    ART=AR
    ISWP=ISWPT
    XLAM=XLAMT
    G8 T8 505
5C9 IF [SW] 5100,5100,900
5100 IF [SW2] 5110,5110,900
5110 IF [ST] 980,980,900
98C LLLL=2
    RETURN
99C LLKK=0
9C2 IZZY = IZZY + 1
    IF [SW] 510,510,901
9C1 CBLAM= COS[CLAMW]/[SIN[CLAMW]]
    ARW=[BW-D]**2/SW
    BCBLAM=BETA*CBLAM
    CR8BT=CR8BTW
    B1=BW
    IAFB=IAFBW
    XMAC=XMACW
    T8VC=T8VCW
    CLAL1=CLALW
    XLAM1=XLAMW
    ISWP1=ISWPW
    BAR=BETA*ARW
    RATI0=CR8BT/[BETA*D]
    LLLL=0
    RETURN
51C IZZY = IZZY + 1
    IF [SW2] 942,942,511
942 LLLL=1
    RETURN
511 CBLAM=COS[CLAMW2]/[SIN[CLAMW2]]
    BCBLAM=BETA*CBLAM
    CR8BT=CR8BW2
    B1=BW2
    IAFB=IAFBW2
    CLAL1=CLALW2
    XLAM1=XLAMW2
    XMAC=XMACW2
    T8VC=T8VCW2
    ISWP1=ISWPW2
    ARW2=[BW2-D]**2/SW2
    BAR=BETA*ARW2
    RATI0=CR8BT/[BETA*D]
    LLLL=0
    RETURN
    END

```



SUBROUTINE CATSUB
 DIMENSION XVXM[16], XDT[16], XAL[48]
 COMMON XVXM, XDT, XAL
 COMMON CN, CA, CKB, CNW, CNT, CNW2, CLW, CLT, CLW2, CL1, CLWB, CLVISW, CLIT,
 1 CLIW, CNTD, CAB, XCP2, XCG2, XCPB, XCPY, XCPW, XCPW2, XCG, XCPTV,
 2 XLAMT4
 COMMON LLKK, ISWPW, ISWPT, ISWPW2, IAFBW, IAFBT, IAFBW2, IL, LLLL, IJ, J,
 1 II, NBSDY, IZZY, ICSC, INASE, NM, NMLK, IDT, IM, IAL, ISWP1, IAFB
 COMMON XLAMW, XLAMT, XLAMW2, XMACW, XMACT, XMACW2, CLAMW, CLAMT, CLAMW2,
 1 BW, BT, BW2, CR0BTW, CR0BT, CR0BW2, SW, ST, SW2, XWING, XTAIL, XWING2
 2 , XL, D, D1, XLN, AREA, XREF, SSUBS, XLBB, ZF, ART, ARW, ARW2, XLNASE
 COMMON C0LAM, BC0LAM, CR0BT, B1, CLAL1, XLAM1, BAR, RAT16, XKTB, XKBT,
 1 XKWB, XKBW, XBCRBW, XKWBW, XKWBT, XKWBW2, XKBBW, XKBTW, XKBW2,
 2 XCPBW, XCPBT, XCPRW2, XCPWB, XCPBT, XCPWB2, XKWBI, BDC, BDCW, BDCT,
 3 BDCW2, CLALW, CLALT, CLALW2, REFT, BETA, AL, TANST, HNST, HT, XKTB1,
 4 VXM, VXMR1, DELTA, XKCRWB, XBCRNB, XMAC, XLAMW4, XLAM24, XLAM2,
 5 XLAM4, T0VC, T0VCT, T0VCW2, T0VCW, EXS, STTB, SWTB, SW2TB, RE,
 6 CD8, CD8W, CDBT, CD8R, CD8W2, CLBW, CLWB2, CLBW2, CLTB, CLBT, CLTD,
 7 CLTDB, CL8DT, CLVIST, CLVIW2, CMB, CD8WBT
 COMMON DC0ESW, DC0B8T, DC0D8P, DC0AW, DC0BT, DC0BW2
 COMMON CDALZ
 XKWB=[2./3*14159]*[[1.+D**4/B1**4]*[.5*ATAN[.5*[B1/D-D/B1]]+3.1415
 19/4.]-[D**2/B1**2]*[[B1/D-D/B1]+2.*ATAN[D/B1]]]/[1.-D/B1]**2
 926 XKTB1=[[3.14159]**2*D**2*[B1/D+1.]**2]/[4.*B1**2]+[3.14159*D**2*[B
 11**2/D**2+1.]**2/[B1**2*[B1/D-1.]**2]]*ARSIN[[B1**2/D**2+1.]/[B1**
 22/D**2+1.]]
 XKTB2=2.*3.14159*D*[B1/D+1.]/[B1*[B1/D+1.]]-[B1**2/D**2+1.]**2/[[
 1B1**2/D**2]*[B1/D-1.]**2]*[ARSIN[[B1**2/D**2-1.]/[B1**2/D**2+1.]]
 2]**2
 XKTB3=[4.*D*[B1/D+1.]/[B1*[B1/D-1.]]]*ARSIN[[B1**2/D**2-1.]/[B1**2
 1/D**2+1.]]-[8./[B1/D-1.]**2]*ALOG[[B1**2/D**2+1.]/[2.*B1/D]]
 XKTB=[1./[3.14159]**2]*[XKTB1-XKTB2-XKTB3]
 IF [ICSC=1] 970, 970, 971
 970 IF [LLKK] 972, 972, 973
 972 XKWB1=D/B1+1.
 Q XKTB1=XKTB
 G0 T0 973
 974 IF [LLKK=1] 973, 973, 972
 973 PAREF=BAR*[1.+XLAM1]*[[1./TBETA*BC0LAM]]+1.
 IF [PAREF=4.] 928, 928, 929
 928 XKWB=[1.+D/B1]**2-XKWB
 G0 T0 1000
 929 IF [IAFB] 9291, 9292, 9291
 9291 IF [BC0LAM=1.] 932, 931, 931
 931 XKWB1=[BC0LAM/[1.+BC0LAM]]*[[[BC0LAM+1.]*[1./RAT19]+BC0LAM]/BC0LA
 1M]**2]*ARCBS[[1.+D1+BC0LAM]*[1./RAT18]]/[BC0LAM*[BC0LAM+1.]*[1./
 2RAT19]]
 XKWB2=[SQR[BC0LAM**2+1.]/[BC0LAM+1.]]*[SQR[1.+2.*BETA*D/CR0BT]*
 11.]-[SQR[BC0LAM**2+1.]/BC0LAM]*[BETA*D/CR0BT]**2*ALOG[[1.+CR0BT/
 2BETA*D]]+SQR[[1.+CR0BT/[BETA*D]]-1.]]-[BC0LAM/[1.+BC0LAM]]*ARCBS
 3[1./BC0LAM]
 XKWB=[8.*BC0LAM/[3.14159*SQR[BC0LAM**2+1.]]*[1.+XLAM1]*[BETA*D/
 1CR0BT]*[B1/D-1.]*[BETA*CLAL1]]*[[XKBW1+XKBW2]]
 G0 T0 1000

932 $XKBW1 = [BCOLAM + (1. + BCOLAM) * \beta D / CR88T] / BCOLAM + 1.5 * [BCOLAM + (1. + BCOLAM) * \beta D / CR88T] / BCOLAM + 5/2$
 $XKBW2 = [(1. + BCOLAM) * \beta D / CR88T] / BCOLAM + 2 * 0.5 * [ALOG[1. + \sqrt{BCOLAM / [BCOLAM + (1. + BCOLAM) * \beta D / CR88T]}]] = ALOG[1. + \sqrt{BCOLAM / [BCOLAM + (1. + BCOLAM) * \beta D / CR88T]}]]$
 $XKBW = [16. * [BCOLAM / (1. + BCOLAM)] + 2 / [3.14159 * (1. + XLAM1) * (\beta D / CR88T) * (B1 / D + 1.) * (\beta CLAL1)]] * [XKBW1 - XKBW2]$
 GO TO 1000
 9292 IF [BCOLAM = 1.] 9295, 9294, 9294
 9294 IF [RATI0 = 1.] 9293, 9311, 9311
 9293 RATI0 = 1.
 $D = CR88T / \beta$
 9311 $XKB1 = [1. + CBLAM * \beta * RATI0] + 2 * \text{ARCOS}[(BCOLAM * RATI0) / (1. + CBLAM * \beta * RATI0)]$
 $XKB2 = BCOLAM * 2 * RATI0 * 2 * \text{ARCS}[1. / BCOLAM] * BCOLAM * RATI0 * 2 * \sqrt{BCOLAM * 2 * 1.} * \text{ARSIN}[1. / RATI0]$
 $XKB3 = \sqrt{BCOLAM * 2 * 1.} * \text{ALOG}[RATI0 + \sqrt{RATI0 * 2 * 1.}]$
 $XKBW = [8. / [3.14159 * \sqrt{BCOLAM * 2 * 1.}] * \beta * CLAL1 * (XLAM1 + 1.) * (B1 / D1 + 1.)] * [1. / RATI0] * [XKB1 - XKB2 - XKB3]$
 GO TO 1000
 9295 IF [RATI0 = 1.] 9296, 9312, 9312
 9296 RATI0 = 1.
 $D = CR88T / \beta$
 9312 $XKB1 = [1. + CBLAM * \beta * RATI0] * \sqrt{[RATI0 + 1.] * [CBLAM * \beta * RATI0 + 1.]}$
 $XKB2 = RATI0 * 2 * [BCOLAM] * 1.5 * BCOLAM * RATI0 * 2 * [BCOLAM + 1.] * [\text{ATAN}[\sqrt{1. / BCOLAM}] - \text{ATAN}[\sqrt{RATI0 - 1.} / (CBLAM * \beta * RATI0 + 1.)]]$
 $XKB3 = [BCOLAM + 1.] / \sqrt{BCOLAM} + 0.5 * [ALOG[1. + \sqrt{BCOLAM * [RATI0 + 1. / (CBLAM * \beta * RATI0 + 1.)]}]] - \text{ALOG}[1. + \sqrt{BCOLAM * [RATI0 - 1.]} / (CBLAM * 2 * \beta * RATI0 + 1.)]]$
 $XKBW = [16. * \sqrt{BCOLAM} * [1. / RATI0] / (\beta * CLAL1 * (XLAM1 + 1.) * (B1 / D1 + 1.) * 3.14159 * [BCOLAM + 1.])] * [XKB1 - XKB2 - XKB3]$
 1000 $XKBT = XKBW - XKTB$
 $D = D1$
 IF [VXM = 1.0] 2000, 2000, 2010
 2000 IF [ISWP1 = 1] 2020, 2020, 2030
 2020 IF [BAR = 2.0] 2040, 2050, 2050
 2040 $XBCRWB = 0.35 * XLAM1 * [0.35 + 17.885 * \sqrt{328.8782 * [BAR = 3.0] * 2}]$
 GO TO 2080
 2050 $XBCRWB = 0.35 - 0.1 * XLAM1$
 GO TO 2080
 2080 IF [BAR = 2.0] 2060, 2070, 2070
 2060 $XCR1 = \sqrt{2029.5 + [BAR = 3.0] * 2}$
 $XBCRWB = [-44.5 + XCR1] - XLAM1 * [-44.5 + XCR1 + 17.885 * \sqrt{328.8782 - [BAR = 3.0] * 2}]$
 GO TO 2110
 2070 $XBCRWB = 0.55 - 0.3 * XLAM1$
 GO TO 2110
 2080 IF [BAR = 4.0] 2090, 2100, 2100
 2090 $XBCRWB = 0.25 * XLAM1 * [32.125 * \sqrt{1032.02 * [BAR = 4.0] * 2}] + [1. - XLAM1] * [ALOG[1.04 + 0.1 * D / B1] * [-7.5 + \sqrt{72.25 * [BAR = 4.0] * 2}]]$
 GO TO 2400
 2100 $XBCRWB = 0.25 + [1. - XLAM1] * ALOG[1.04 + 0.1 * D / B1]$
 GO TO 2400

211C IF [BAR=4.0] 2120,2130,2130
 212C XBCRWB=0.25*[1.0+XLAM1]*[32.125-SQRT[1032.02-[BAR=4.0]**2]]*
 1*[1.0+XLAM1]*AL0G[1.12+0.3*D/B1]*[-7.5+SQRT[72.25-[BAR=4.0]**2]]
 GO TO 2400
 213C XBCRWB=0.25*[1.0+XLAM1]*AL0G[1.12+0.3*D/B1]
 GO TO 2400
 201C IF [ISWP1=1] 2140,2140,2150
 214C IF [BAR=3.0] 2160,2170,2170
 216C XBCRWB=-[9.235+25.0*[1.0+XLAM1]]+SQRT[[9.71+25.0*[1.0+XLAM1]]**2-
 1*[BAR=3.0]**2]
 GO TO 2200
 217C XBCRWB=0.005*BAR+0.46
 GO TO 2200
 215C IF [BAR=3.0] 2180,2190,2190
 218C XBCRWB=0.675-XLAM1*[0.675+9.235-SQRT[94.1-[BAR=3.0]**2]]
 GO TO 2300
 219C XBCRWB=0.005*BAR+0.46+0.2*[1.0+XLAM1]
 GO TO 2300
 220C BARLAM=BAR*[1.0+XLAM1]*[1.0+[1.0/BCBLAM]]
 IF [BARLAM=4.0] 2210,2210,2220
 221C IF [BAR=2.0] 2230,2240,2240
 223C XBMID1=AL0G[1.32-0.32*XLAM1]
 2235 XBMID2=4.0+[XBMID1**2]-[[0.5+0.5139*[D/B1]*[1.17+XLAM1]*[1.0/[0.331
 1**[D/B1]]]]**2]
 XBMID3=2.0+[[XBMID1]+[0.5+0.5139*[D/B1]*[1.17+XLAM1]*[1.0/[0.331
 1**[D/B1]]]]]
 XBMID=XBMID2/XBMID3
 XCMID=4.0+[XBMID1*XBMID]**2
 XBCRWB=SQRT[XCMID-[BAR=2.0]**2]+XBMID
 GO TO 2400
 224C XBCRWB=0.5+0.25695*[D/B1]*[1.0/[0.331**[D/B1]]]*BAR*[1.17+XLAM1]
 GO TO 2400
 222C IF [IAFB] 2250,2250,2260
 225C IF [RAT10=1.0] 2280,2280,2270
 228C XBCRWB=0.67
 GO TO 2400
 227C XBCRWB=-2.32+SQRT[8.9401-[(1.0/RAT10)-1.0]**2]
 GO TO 2400
 226C XBCRWB=[0.429/RAT10]+0.5
 GO TO 2400
 230C BARLAM=BAR*[1.0+XLAM1]*[1.0+[1.0/BCBLAM]]
 IF [BARLAM=4.0] 2310,2310,2220
 231C IF [BAR=2.0] 2320,2240,2240
 232C XBMID1=AL0G[1.65-0.65*XLAM1]
 GO TO 2235
 240C D=D1
 ARAT=BAR/BETA
 IF [ARAT=3.0] 20,20,21
 21 ARAT=3.0
 20 PDC=2.0*[1.0+XLAM1]**1.6*EXP[-4*ARAT]
 IF [ARAT=1.0] 22,23,23
 23 PDC=PDC-0.25*[ARAT=1.0]
 22 CONTINUE
 RETURN
 END

TABLE 2

INPUT NOMENCLATURE*

AREA	arbitrary reference area, ft ²
BT	total tip-to-tip tail span including the missile body**
BW	total tip-to-tip wing span including the missile body
BW2	total tip-to-tip wing two span including the missile body
CLAMT	leading edge sweep angle of the tail***
CLAMW	leading edge sweep angle of the wing
CLAMW2	leading edge sweep angle of wing two
CROOTT	tail root chord
CROOTW	wing root chord
CROOTW2	wing two root chord
D	body diameter
HT	altitude in feet
IAFBT	afterbody constant for the tail 0 - no afterbody following the tail 1 - afterbody following the tail
IAFBW	afterbody constant for the wing 0 - no afterbody following the wing 1 - afterbody following the wing
IAFW2	afterbody constant for wing two 0 - no afterbody following wing two 1 - afterbody following wing two
IAL	number of angles of attack
ICSC	control surface constant 1 - tail control 2 - wing control 3 - canard control
IDT	number of control surface deflection angles (must beat least one; if there is no control surface, IDT = 1, DELTA = 0.0)
IM	number of Mach numbers

TABLE 2
(continued)

INOSE	nose constant 1 - blunted ogive or cone 2 - pointed ogive 3 - pointed cone
ISWPPT	sweep constant of tail 1 - unswept mid-chord 2 - unswept trailing edge
ISWPW	sweep constant of wing 1 - unswept mid-chord 2 - unswept trailing edge
ISWPW2	sweep constant of wing two 1 - unswept mid-chord 2 - unswept trailing edge
NBODY	number of configurations being run (a configuration is one complete data deck)
ST	exposed planform area of one pair of tail panels
SW	exposed planform area of one pair of wing panels
SW2	exposed planform area of one pair of wing two panels
TOVCT	thickness-to-chord ratio of the tail
TOVCW	thickness-to-chord ratio of the wing
TOVCW2	thickness-to-chord ratio of wing two
XAL	missile angle of attack (degrees)
XCG	missile center of gravity location as measured from the nose
XDT	control surface deflection angle (degrees)
XL	missile length
XLAMT	tip-to-root-chord ratio of the tail
XLAMW	tip-to-root-chord ratio of the wing
XLAMW2	tip-to-root-chord ratio of wing two
XLNOSE	length of the nose

TABLE 2
(continued)

XMACT	mean geometric chord of the tail
XMACW	mean geometric chord of the wing
YMACW2	mean geometric chord of wing two
XREF	arbitrary reference length
XTAIL	distance from the nose to the leading edge of the tail root chord
XVXM	missile flight Mach number
XWING	distance from the nose to the leading edge of the wing root chord
XWING2	distance from the nose to the leading edge of wing two root chord

* The control surface is defined as the tail regardless of the mode of control, and the fixed surface(s) is (are) always defined as the wing(s). See Figure 1.

** All linear dimensions are in feet.

***All angular dimensions are in degrees.

TABLE 3
PROGRAM INPUT FORMAT

1	Run Identification	Configuration Title and/or Number (1st 60 columns)																
2	Control Constants	ICSC (15)	INOSE (15)	IDT (15)	IM (15)	IAL (15)	NBODY (15)											
3	Wing Inputs	ISWPW (15)	IAFBW (15)	XLAMW (F10.5)	CLAMW (F10.5)	BW (F10.5)	CROOTW (F10.5)	SW (F10.5)	XMACW (F10.5)	XWING (F10.5)								
4	Wing 2 Inputs	ISWPW2 (15)	IAFBW2 (15)	XLAMW2 (F10.5)	CLAMW2 (F10.5)	BW2 (F10.5)	CROOTW2 (F10.5)	SW2 (F10.5)	XMACW2 (F10.5)	XWING2 (F10.5)								
5	Tail Inputs	ISWPWT (15)	IAFBT (15)	XLAMT (F10.5)	CLAMT (F10.5)	BT (F10.5)	CROOTT (F10.5)	ST (F10.5)	XMACT (F10.5)	XTAIL (F10.5)								
6	Miscellaneous Data	HT (F10.3)	D (F10.3)	XL (F10.3)	XLNOSE (F10.3)	XCG (F10.3)	AREA (F10.3)	XREF (F10.3)										
7	Miscellaneous Data	TOVCW (F10.3)	TOVCW2 (F10.3)	TOVCT (F10.3)														
8	Control Surface Deflection Angles	XDT (F5.1)	Any number of deflection angles up to 16 may be input.															
9	Missile Flight Mach Numbers	XVXM (F5.1)	Any number of Mach numbers up to 16 may be input.															
10	Missile Angles of Attack	XAL (F5.1)	Any number of angles of attack up to 48 may be input.															
11	Missile Angles of Attack	If this card is not required, leave out of data deck.																
12	Missile Angles of Attack	If this card is not required, leave out of data deck.																

TABLE 4
OUTPUT NOMENCLATURE

AL	missile angle of attack, degrees
CA	total axial force coefficient
CDTOT	total missile drag coefficient
CLALT	$C_{L\alpha}$ of the tail
CLALW	$C_{L\alpha}$ of the wing
CLALW2	$C_{L\alpha}$ of wing two
CLB	body lift coefficient
CLI	lift loss due to downwash
CLTOT	total missile lift coefficient
CLTT	tail lift coefficient
CLWT	wing lift coefficient
CM	total pitching moment coefficient about the missile center of gravity
CN	total normal force coefficient
CNB	body normal force coefficient
CNT	tail normal force coefficient
CNTD	tail normal force coefficient due to control surface deflection
CNW	wing normal force coefficient
DELTA	control surface deflection angle, degrees
VXM	missile flight Mach number
XCPB	body center of pressure location as measured from the nose, feet
XCPT	tail center of pressure location as measured from the nose, feet

$$CMB = CNB(XCG - XCPB)$$

TABLE 4
(continued)

XCPW wing center of pressure location as measured from the nose,
feet

XCP2 total missile center of pressure location as measured from
the nose, feet

XCP ~~XCPW XCP2~~

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13. ABSTRACT

A method for predicting the static, longitudinal aerodynamic characteristics of typical missile configurations at zero roll angle (i.e., in a plus configuration) has been developed and programmed for use on the IBM 7090 digital computer. It can be applied throughout the subsonic, transonic, and supersonic speed regimes to slender bodies of revolution or to nose-cylinder body combinations with low aspect-ratio lifting surfaces. The aerodynamic characteristics can be computed for missile configurations operating at angles of attack up to 180 degrees. The effect of control surface deflections for all modes of aerodynamic control are taken into account by this method. The method is based on well-known linear, nonlinear crossflow and slender body theories with empirical modifications to provide the high angle of attack capability. Comparisons of the theory with experimental data are presented to demonstrate the accuracy of the method.

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